Next Assembly	Used On
First Application	

REVISIONS

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MC 023862

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E1	Revise per ECN/TRP2395, CCR 1573. OAB-VD810-0007-01	CM 2/22/99	2/22/99
E2	Revise per ECN/TRP1890, CCR 2337 OAB-VD810-0014-01	CM 2/22/99	2/22/99
F	Incorporated per TRP1890, TRP2395 OAB-VL810-0007-01X; CCR 1573	CM 2/22/99	2/22/99
F1	Revised per EJA/ECN TRP2134 CCR 1627, CCR 2353, CD 615, Mod 348, and CD 621, Mod 351. OAV-VD810-0019-01. CM	3/1/00	L. Roberts
G	Incorporate EJA/ECN TRP2134. Revise per GSFC draft task order dtd. December 1999. Wang to Word conversion, typos, LM SV format. Reference TRP 2657. OAB-VL810-0020-01X. CM	3/1/00	L. Roberts
H	Revised & Incorporated per EJA/ECN TRP2643, CCR 2393A OAB-VL810-0036-01X. L. Roberts	8/17/00	8/17/00
J	Revised & Incorporated per ECN TRP 3091; CCR 1874, Mod 474; QAB-VL810-0059-01X.	2/5/02	04 FEB 02

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CM 03-28-95	Written Susan Gregg Date 16 February 1995	UNIQUE INSTRUMENT INTERFACE SPECIFICATION FOR THE MICROWAVE HUMIDITY SOUNDER (MHS)	
	Approved Susan Gregg Date 16 February 1995		
PA/QA 2/16/95	Approved R.M. Cummings Date 2/16/95	Size A	Code Ident No. 18713 — IS 20046415
Adams 3/3/95	DPC 9313969P.DOC		Sheet 1 of 99

869 7/93

TIROS

UNIQUE INTERFACE SPECIFICATION

FOR THE

MICROWAVE HUMIDITY

SOUNDER (MHS)

Approval:

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EUMETSAT: Instrument Project Manager

Spacecraft Contractor: Systems Manager

EUMETSAT: EPS Programme Manager

Spacecraft Contractor: TIROS Program Manager

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ITAR CONTROLLED DATA

Size	Code Ident No.	
A	18713	IS20046415
		Sheet 2

772 6/93

TABLE OF CONTENTS

TABLE OF CONTENTS (CONTINUED)

TABLE OF CONTENTS (CONTINUED)

TABLE OF CONTENTS (CONTINUED)

LIST OF APPENDICES

LIST OF ILLUSTRATIONS

LIST OF TABLES

1.0 SCOPE

This document establishes the electrical, mechanical and thermal interfaces between the Microwave Humidity Sounder (MHS) and the Advanced TIROS-N Spacecraft (ATN) and Spacecraft Aerospace Ground Equipment.

This document details all environments which will be seen by the instrument from the time of its arrival at the Spacecraft Contractor Facility through spacecraft launch, including all phases of storage and test. In addition, specific information about unique instrument properties or requirements in associated areas (such as power and handling requirements, test requirements, test equipment, targets, etc.) is contained herein.

2.0 DOCUMENTS

2.1 Applicable Documents

The current issues of the following documents relate to the interface. Changes to these documents which affect form or function of the spacecraft interface will be submitted to the NASA TIROS Project Office for CCR Action.

2.1.1 Martin Marietta Originated Documents

(1)	3278778	Field of View Drawing
(2)	3287774	KLM RSS Thermal Finishes
(3)	3287775	KLM ESM Thermal Finishes
(4)	3287776	KLM IMP Thermal Finishes
(5)	3278200	ATN Spacecraft Assembly
(6)	3278776	ATN-KLM ESM Assembly, GFE
(7)	20041550	MHS Configuration Drawing
(8)	20057860	MHS/EM-MIU ETM Compatibility Test Plan

2.1.2 Instrument Contractor Originated Documents

(1)	MHS-ID-GA002-MMP	Thermal Interface Control Drawing
(2)	MHS-TN-JA097-MMP	Reduced Thermal Mathematical Model Description
(3)	MHS-ID-JA029-MMP	MHS Instrument Configuration
(4)	MHS-ID-FA002-MMP	MHS Instrument Footprint ICD
(5)	MHS-ID-JA046-MMP	MHS Alignment ICD
(6)	MHS-ID-JA071-MMP	MHS Instrument Field-of-View
(7)	MHS-IS-JA035-MMP	(MHS Electrical ICD)

(8)	3175-35025-29-1	Drill Template Drawing
(9)	MHS-TN-JA090-MMP	Finite Element Model Description
(10)	MHS-TN-JA271-MMP	MHS Integration Constraints Document*
(11)	MHS-OM-JA215-MMP	MHS Flight Operations Manual*
(12a)	3175-34024-MAU	EGSE User Manual
(12b)	3175-35032-MAU	Lifting and Handling Devices User Manual
(12c)	3175-35042-MAU	Cryostat User Manual
(12d)	3175-35043-MAU	Transportation Container
(12e)	MHS104 (M/RSI/41/5/2) Issue 2	Special Calibration Test Equipment (SCTE) User Guide
(13)	TBS	MHS Spacecraft Level Test Manual*
(14)	MHS-TN-JA063-MMP-Issue 5	MHS TM-TC and Science Data Format Directory
(15)	3175-JA001-01-0	MHS General Assembly(GA) Drawing

2.1.3 NASA Originated Documents

(1)	GSFC-S-480-72	Command/Telemetry Bus General Specification
(2)	GSFC-S-480-73	Low Rate Bus General Specification
(3)	GSFC PPL-19, October 1989 (Rev. 1, 3/01/91) - Spacecraft	
(4)	MIL-M-38510/104B, 29 May 1987	Military Specification, Microcircuits, Linear Line Drivers` and Receivers, Monolithic Silicon
(5)	GSFC S-480-53	Instrument Interface Description for NOAA 2000 Instruments with European Morning Spacecraft

2.1.4 EUMETSAT Originated Documents

(1)	EPS/MHS/SPE/93001	Performance and Functional Specification for the Microwave Humidity Sounder
(2)	EPS/MHS/REQ/93001	MHS Product Assurance Requirements

* 3 Months Prior to delivery of flight models to the Spacecraft Contractor (Final Calibration Data will not be available until 2 weeks prior to delivery).

3.0 REQUIREMENTS

3.1 Electrical

3.1.1 Grounds

The instrument will conform to the grounding scheme detailed in Figure 1; however, signal shields which come from the spacecraft harness, and which are grounded on the spacecraft, will not be tied to instrument case ground.

There will be a ground strap provided by the Spacecraft Contractor. Matra Marconi will provide the ground strap attach point on the instrument.

There will be an overall cable shield for each connector. This will be grounded to the connector shell at each end of the MIU/MHS spacecraft cable.

3.1.2 Connectors

a) General. Instruments shall use connectors to interface with the spacecraft harness. Connectors shall be selected in accordance with the applicable documents and shall be limited to those that accept #20 AWG wire. Instrument connectors shall provide grounding to the shells of mating connectors.

b) Power. Power inputs and their respective returns shall be on the same connector of the instrument. The power/power return connector shall utilize pins. Each circuit shall be conveyed on a minimum of two pins wired in parallel. A pin designated "chassis ground" which is internally connected to the instrument housing shall be provided.

c) Signals and Data. Signals and data shall be on connector(s) other than those provided for power. These connectors shall be female connectors. At least one connector shall provide a connection designated "chassis ground" which is internally connected to the instrument housing chassis grounding point. Use of this connection in the spacecraft ground will be coordinated between the respective parties. Each signal/data connector shall provide a pin designated "signal ground" which is internally connected to the signal ground for providing reference levels to external electronic equipments. Bi-Polar (two wire) signals shall use connections which are located adjacent to each other. Connections which convey shields shall be located adjacent to the connections which convey the signals which require the shields.

d) Intra-instrument Signals. When an instrument is comprised of more than one cased assembly then intra-instrument signals shall be conveyed by connectors reserved for that purpose.

e) GSE Access Connectors. If an instrument must be accessed by GSE for test purposes while installed in the spacecraft, connectors reserved for that purpose shall be provided.

f) Screws used for holding the spacecraft harness connectors to the MHS instrument connectors shall be torqued into their respective jackposts at a value of 2.5 ± 0.3 in.-lb.

3.1.2.1 Connector Allocation

Connector requirements for the instrument shall be as shown in Table 1.

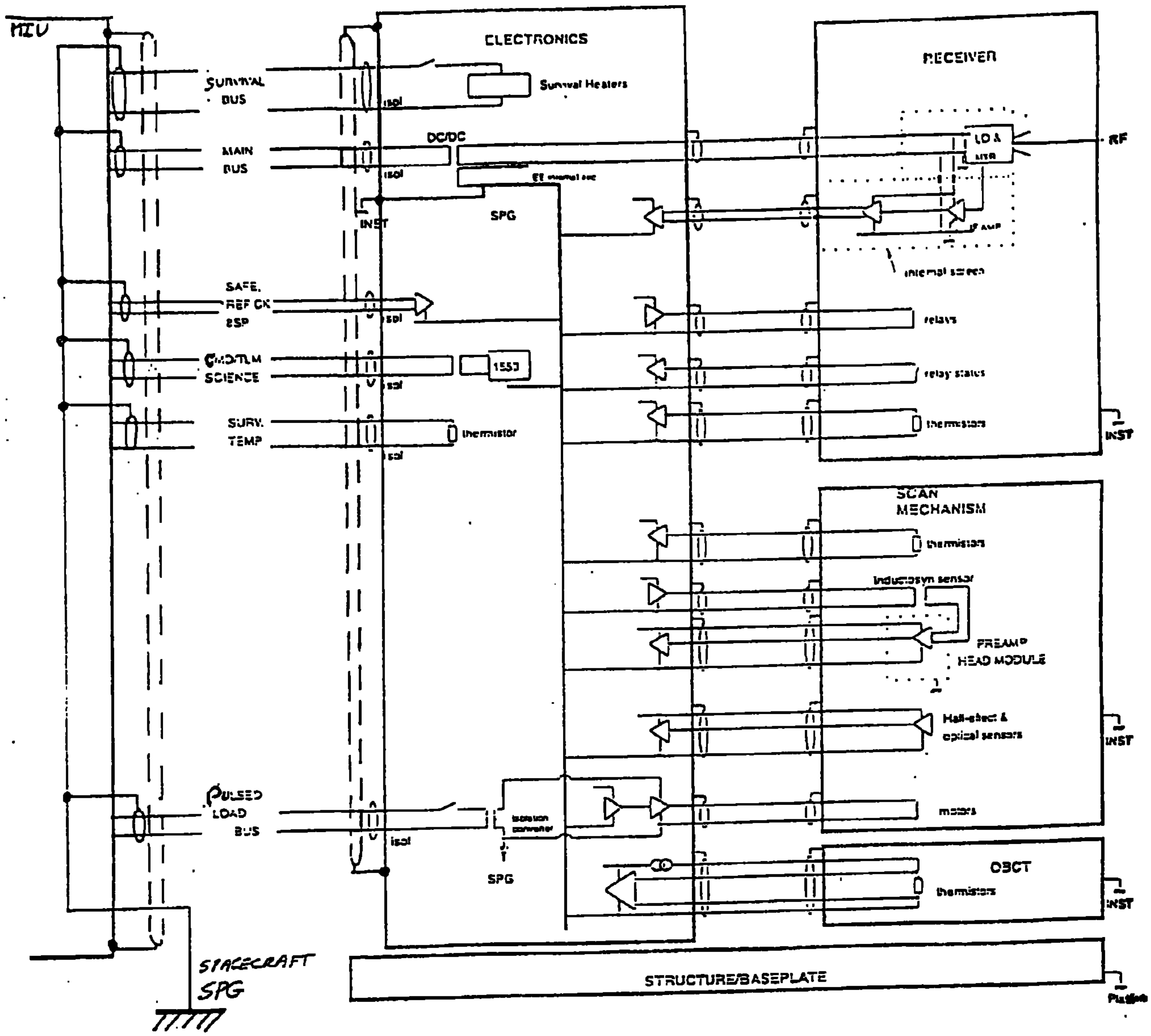


Figure 1. MHS Grounding and Return Line Diagram

ITAR CONTROLLED DATA

Size	Code Ident No.	
A	18713	IS-20046415
		Sheet 13

TABLE 1. MHS INSTRUMENT CONNECTOR REQUIREMENTS

Reference	Type	Function
J100	D37P	Primary Power (Main/Pulse Load/Survival)
J101	D78S	Differential Interfaces A (8SP/REF CK/SAFE A)
J102	D78S	Differential Interfaces B (8SP/REF CK/SAFE B)
J103	D15S	Discrete TM/TC (Survival Temperature TM, Main Bus Select TM, Converter Protection Disable TC)
J104	D9S	1553 Cmd/Tlm Bus A & B
J105	D9S	1553 Cmd/Tlm Bus A & B
J106	D9S	1553 Science Data A & B
J107	D9S	1553 Science Data A & B
J108	D9S	1553 Cmd/Tlm Bus A & B Address
J109	D9S	1553 Cmd/Tlm Bus A & B Address
J110	D9S	1553 Science Bus A & B Address
J111	D9S	1553 Science Bus A & B Address
J192	D9P	Survival Heater Power (for Safety Supply)
Connector Type is the type mounted on the instrument. DxxS/P: Subminiature D-Type, xx way Socket/Plug. All the above connectors are mounted on the -A (+Z) face of the electronics equipment. Note: There is no keying of connectors.		

3.1.2.2 Connector Keying Requirements

The connector keying will not be provided. Improper mating of the spacecraft connectors shall not damage the instrument.

3.1.2.3 Harness Mating Connectors

The mating connector requirements for the spacecraft harness shall be as shown in Table 2.

3.1.2.4 Pin Designations

Connector pin designations and shielding requirements for the instrument interface connectors and the mating spacecraft harness shall be as shown in Table 3.

3.1.2.5 Intra-Instrument Harness Requirements

NONE

3.1.2.6 Connector Location and Access

The interface connectors to the spacecraft harness shall be located on the +Z (-A, Sun Side) face of the instrument, and shall not extend beyond the instrument envelope.

3.1.3 Power

3.1.3.1 Power Sources

(1) The main power required by the MHS instrument shall be taken from the +28-Volt Main Bus.

(2) The +28-Volt Pulse Load Bus shall be used to supply power to the motors and operational heaters in the MHS.

(3) A separate +28-Volt Pulse Load Bus shall be used to supply power to the survival heater. The MIU will provide protection diodes on the survival heater bus (because the survival heaters will also be used as safety heaters during spacecraft thermal vacuum testing).

(4) The power drawn from the above sources will be as shown in Table 4. The total power from all the buses shall not exceed 95.0 watts worst-case end of life orbit average.

(5) Switching will be done by the spacecraft and not by the instrument.

(6) The spacecraft shall provide hookup and wiring with the following impedance characteristics between the instrument power terminals and the power source. For frequencies below 1 MHz, the hookup and powerline impedance for each power line wire and each return wire may be represented by a resistor (R1) in series with an Inductor (L1), with this series combination shunted by a resistor (R2). The limiting values of the components are ≤ 0.3 ohms for R1, ≤5 microhenries for L1, and ≤30 ohms for R2.

TABLE 2. SPACECRAFT HARNESS CONNECTOR REQUIREMENTS

S/C Designation	Martin Marietta Part No.	Type	Description (Pins, Sex)	Function
J100	1721489-4	D37S	37, Female	Primary Power
J101		D78P	78, Male	Differential Interface A
J102		D78P	78, Male	Differential Interface B
J103	1721490-2	D15P	15, Male	Discrete TM/TC
J104	1721490-1	D9P	9, Male	1553 CMD/TLM Bus A & B
J105	1721490-1	D9P	9, Male	1553 CMD/TLM Bus A & B
J106	1721490-1	D9P	9, Male	1553 Science Data Bus A & B
J107	1721490-1	D9P	9, Male	1553 Science Data Bus A & B
J108	1721490-1	D9P	9, Male	1553 CMD/TLM Bus A & B Address
J109	1721490-1	D9P	9, Male	1553 CMD/TLM Bus A & B Address
J110	1721490-1	D9P	9, Male	1553 Science Bus A & B Address
J111	1721490-1	D9P	9, Male	1553 Science Bus A & B Address
J192	1721489-1	D9S	9, Female	Survival Heater Power

TABLE 3A. CONNECTOR PIN DESIGNATIONS

Connector: J100 Power

Spacecraft	MHS
P100	J100 (D37P)

Pin Number	Signal Name	Pin Number	Signal Name
1	EQT Chassis	29	Pulse Pwr A RTN
2	Main Pwr A	11	Pulse Pwr A RTN
21	Main Pwr A	30	Pulse Pwr A RTN
3	Main Pwr A	12	Pulse Pwr B RTN
22	Main Pwr A RTN	31	Pulse Pwr B RTN
4	Main Pwr A RTN	13	Pulse Pwr B RTN
23	Main Pwr A RTN	32	Pulse Pwr B
5	Main Pwr B RTN	14	Pulse Pwr B
24	Main Pwr B RTN	33	Pulse Pwr B
6	Main Pwr B RTN	Shell	Pulse Pwr Screen (Shield)
25	Main Pwr B	16	Survival Pwr
7	Main Pwr B	35	Survival Pwr
26	Main Pwr B	17	Survival Pwr
Shell	Main Pwr Screen (Shield)	36	Survival Pwr RTN
9	Pulse Pwr A	18	Survival Pwr RTN
28	Pulse Pwr A	37	Survival Pwr RTN
10	Pulse Pwr A	Shell	Survival Power Screen (Shield)
8, 15, 19, 20, 27, 34 connected to chassis in equipment			

TABLE 3B. CONNECTOR PIN DESIGNATIONS

Connector: J101 Clock/Sync

Spacecraft	MHS
P101	J101 (D78S)

Pin Number	Signal Name
59	REF CLOCK A +
39	REF CLOCK A -
Shell	REF CLOCK A Screen (Shield)
58	SAFE A +
38	SAFE A -
Shell	SAFE A Screen (Shield)
57	8 Sec Sync Pulse A +
37	8 Sec Sync Pulse A -
Shell	8 Sec Sync Pulse A Screen (Shield)
Others	Reserved – no connection in harness

TABLE 3C. CONNECTOR PIN DESIGNATIONS

Connector: J102 Clock/Sync

Spacecraft	MHS
P102	J102 (D78S)

Pin Number	Signal Name
59	REF CLOCK B +
39	REF CLOCK B -
Shell	REF CLOCK B Screen (Shield)
58	SAFE B +
38	SAFE B -
Shell	SAFE B Screen (Shield)
57	8 Sec Sync Pulse B +
37	8 Sec Sync Pulse B -
Shell	8 Sec Sync Pulse B Screen (Shield)
Others	Reserved – no connection in harness

TABLE 3D. CONNECTOR PIN DESIGNATIONS

Connector: J103 Discrete TM/TC

Spacecraft	MHS
P103	J103 (D15S)

Pin Number	Signal Name
2	Survival Temp EE +
9	Survival Temp EE -
Shell	Survival Temp EE Screen (Shield)
3	Survival Temp RX +
10	Survival Temp RX -
Shell	Survival Temp RX Screen (Shield)
4	Survival Temp SM +
11	Survival Temp SM -
Shell	Survival Temp SM Screen (Shield)
8	Main Bus Select Status +
15	Main Bus Select Status -
Shell	Main Bus Select Status Screen (Shield)
6	Main CV PROT Disable +
13	Main CV PROT Disable -
7	RF CV PROT Disable +
14	RF CV PROT Disable -
Shell	CV Disables Screen (Shield)
1, 5, 12	EQT Chassis

TABLE 3E. CONNECTOR PIN DESIGNATIONS

Connector: J104 1553 Bus

Spacecraft	MHS
P104	J104 (D9S)

Pin Number	Signal Name
1	Cmd/Tlm Bus A +
6	Cmd/Tlm Bus A -
Shell	Cmd/Tlm Bus A Screen (Shield)
5	Cmd/Tlm Bus B +
9	Cmd/Tlm Bus B -
Shell	Cmd/Tlm Bus B Screen (Shield)
Others	Floating (See note)

Note: Floating pins may be connected to connector shell on spacecraft side.

TABLE 3F. CONNECTOR PIN DESIGNATIONS

Connector: J105 1553 Bus

Spacecraft	MHS
P105	J105 (D9S)

Pin Number	Signal Name
1	Cmd/Tlm Bus A +
6	Cmd/Tlm Bus A -
Shell	Cmd/Tlm Bus A Screen (Shield)
5	Cmd/Tlm Bus B +
9	Cmd/Tlm Bus B -
Shell	Cmd/Tlm Bus B Screen (Shield)
Others	Floating (See J104 note)

TABLE 3G. CONNECTOR PIN DESIGNATIONS

Connector: J106 1553 Bus

Spacecraft	MHS
P106	J106 (D9S)

Pin Number	Signal Name
1	Science Data Bus A +
6	Science Data Bus A -
Shell	Science Data Bus A Screen (Shield)
5	Science Data Bus B +
9	Science Data Bus B -
Shell	Science Data Bus B Screen (Shield)
Others	Floating (See J104 note)

TABLE 3H. CONNECTOR PIN DESIGNATIONS

Connector: J107 1553 Bus

Spacecraft	MHS
P107	J107 (D9S)

Pin Number	Signal Name
1	Science Data Bus A +
6	Science Data Bus A -
Shell	Science Data Bus A Screen (Shield)
5	Science Data Bus B +
9	Science Data Bus B -
Shell	Science Data Bus B Screen (Shield)
Others	Floating (See J104 note)

TABLE 3I CONNECTOR PIN DESIGNATIONS

Connector: J108 1553 Bus Address (MHS Side A)

Spacecraft	MHS
P108	J108 (D9S)

Pin Number	Signal Name
1	Cmd/Tlm Bus ADDR A4
2	Cmd/Tlm Bus ADDR A3
3	Cmd/Tlm Bus ADDR A2
4	Cmd/Tlm Bus ADDR A1
5	Cmd/Tlm Bus ADDR A0
6	Cmd/Tlm Bus ADDR Parity
7	Signal 0V
8	Signal 0V
9	Floating (See notes)

Notes:
- This connector shall be wired to address 16 decimal. Sides A and B will have the same address.
- Floating pin may be connected to connector shell on spacecraft side but not to "signal 0V".

TABLE 3J. CONNECTOR PIN DESIGNATIONS

Connector: J109 1553 Bus Address (MHS Side B)

Spacecraft	MHS
P109	J109 (D9S)

Pin Number	Signal Name
1	Cmd/Tlm Bus ADDR A4
2	Cmd/Tlm Bus ADDR A3
3	Cmd/Tlm Bus ADDR A2
4	Cmd/Tlm Bus ADDR A1
5	Cmd/Tlm Bus ADDR A0
6	Cmd/Tlm Bus ADDR Parity
7	Signal 0V
8	Signal 0V
9	Floating (See J108 note)

Note: This connector shall be wired to address 16 decimal. Sides A and B will have the same address.

TABLE 3K. CONNECTOR PIN DESIGNATIONS

Connector: J110 1553 Bus Address (MHS Side A)

Spacecraft MHS

P110 J110 (D9S)

Pin Number	Signal Name
1	Science Data Bus ADDR A4
2	Science Data Bus ADDR A3
3	Science Data Bus ADDR A2
4	Science Data Bus ADDR A1
5	Science Data Bus ADDR A0
6	Science Data Bus ADDR Parity
7	Signal 0V
8	Signal 0V
9	Floating (See J108 note)

Note: This connector shall be wired to address 7 decimal. Sides A and B will have the same address.

TABLE 3L. CONNECTOR PIN DESIGNATIONS

Connector: J111 1553 Bus Address (MHS Side B)

Spacecraft — MHS

P111 — J111 (D9S)

Pin Number	Signal Name
1	Science Data Bus ADDR A4
2	Science Data Bus ADDR A3
3	Science Data Bus ADDR A2
4	Science Data Bus ADDR A1
5	Science Data Bus ADDR A0
6	Science Data Bus ADDR Parity
7	Signal 0V
8	Signal 0V
9	Floating (See J108 note)

Note: This connector shall be wired to address 7 decimal. Sides A and B will have the same address.

TABLE 3M. CONNECTOR PIN DESIGNATIONS

Connector: J192 Survival Heaters

Spacecraft	MHS
P192	J192 (D9P)

Pin Number	Signal Name
6	Surv. HTRS PWR (Safety)
7	Surv. HTRS PWR (Safety)
1	Surv. HTRS PWR (Safety) RTN
2	Surv. HTRS PWR (Safety) RTN
Shell	Surv. HTRS PWR (Safety) Screen (Shield)
3, 4, 5, 8, 9	EQT Chassis

TABLE 4. MHS POWER REQUIREMENTS

+28V Main Bus		+28V Pulse Load Bus		+28V Pulse Load Bus (Survival Heater)	
Avg. (watts)	Current (amp)	Avg. (watts)(1)	Current (amp)(2)	Avg. (watts)	Current (amp)
84.0	3.5	50.0	3.0	66	3.0

NOTE: Powers are orbit average for the normal operating (scan) mode.
The MHS orbit average power on the combined +28V Main and Pulse Load Buses shall not exceed 95.0 watts. MHS survival heater is only used during survival conditions in which the MHS is not operating.

(1) Maximum power is during Power-On/Warm-Up Modes due to Operational Heaters
(2) Maximum peak current is in Scan Mode due to motor current profile.

Currents (amps) are maximum steady-state for the +28V Main Bus and the +28V PLB (survival). Currents are peak values for the +28V Pulse Load Bus due to motor currents.

Mode	Description	+28V Main Bus Average Power (Watts)
Launch/Off	Instrument off	0.00
Power-On	EE on, Rx and SM warming up to start-up temperatures	45.0
Warm-Up	Rx on, SM on (motors idle) EQTS warming up to full operating temperature	84.0
Standby	All EQTS on and at full operating temperatures	84.0
Scan Mode	Nominal operating mode, collecting science data	84.0
Fixed View Mode	Operating mode with reflector at fixed target, collecting science data	84.0
Self-Test Mode	Performing special test functions	45.0/84.0
Safeing Mode	Parking reflector and switching off Rx and SM, in preparation of removal of power	84.0
Fault Mode	Entered when an anomaly is detected in telemetry or instrument behavior	45.0

Note: In any single mode the maximum combined load on the +28V Main and Pulse Load Buses shall not exceed 95.0 W.

TABLE 4. MHS POWER REQUIREMENTS (Continued)

Mode	Description	+28V Pulse Load Bus Average Power (Watts)	+28V Pulse Load Bus Average Power (Survival Heater) (Watts)
Launch/Off	Instrument off	0.0	66.0
Power-On	EE on. Rx and SM warming up to start-up temperatures	50.0	0
Warm-Up	Rx on, SM on (motors idle) EQTS warming up to full operating temperature	50.0	0
Standby	All EQTS on and at full operating temperatures	7.0	0
Scan Mode	Nominal operating mode, collecting science data	12.0	0
Fixed View Mode	Operating mode with reflector at fixed target, collecting science data	7.0[1]	0
Self-Test Mode	Performing special test functions	50.0	0
Safeing Mode	Parking reflector and switching off Rx and SM, in preparation of removal of power	12.0	0
Fault Mode	Entered when an anomaly is detected in telemetry or instrument behavior	50.0	0

1. Exclude transients due to change of Fixed View positions.

Note: In any single mode the maximum combined load on the +28V Main and Pulse Load Buses shall not exceed 95.0W.

EE = Electronics Equipment
Rx = Receiver
SM = Scan Mechanism
EQTS = Equipments

3.1.3.2 +28-Volt Main Bus Power Requirements

The spacecraft will provide regulated power at a level of +28.0 ±0.56 volts at the PSE, exclusive of ripple and transients described in Items 2, 3 and 4. The PSE regulated main bus power is input to the MIU which then provides regulated power to the MHS. Due to a voltage drop internal to the MIU, the main bus voltage provided to the MHS will be 28.56 volts maximum and 26.94 volts minimum. The voltage on the +28 Volt Main Bus will not exceed +38.0 volts or drop below +16.0 volts, including all ripple and transients.

(1) ***Source Impedance.*** The small signal source impedance of the +28 Volt Main Bus at the instrument will not exceed 0.3 ohms at all frequencies below 100 kHz. This assumes 0.1 ohms for fuse and line losses.

(2) ***Voltage Ripple.*** Sinusoidal voltage ripple, including repetitive spikes, on the +28 Volt Main Bus will not exceed 200 millivolts peak-to-peak for frequencies from DC to 100 kHz, including load current ripple as specified in Paragraph 3.1.3.2.3.

(3) ***Source Failure Mode Transients.*** The instrument shall survive, without permanent degradation, the following transients which may appear on the +28-volt Main Bus. These transients may appear when the spacecraft supply switches to a backup regulator in the event of a failure of the primary regulator. The instrument shall return to normal operation within one complete operational cycle after the bus has returned to its regulation band.

(a) ***Under-Voltage Behavior.*** A voltage on the +28-volt Main Bus of less than +27 volts and more than +16 volts will constitute an under-voltage condition. This condition may exist for a period of up to 3.0 scconds.

(b) ***Over-Voltage Behavior.*** A voltage on the +28-volt Main Bus of greater than 29.50 volts will constitute an over-voltage condition. Voltages greater than 30.5 volts will exist for no longer than 50 ms and will be immediately followed by an under-voltage condition for up to 1.5 seconds.

(4) ***Source Voltage Transients (Operational).*** The instrument shall continue to operate within specification when the following transients appear on the +28-volt Main Bus.

(a) ***Low Frequency Load-Induced Turn-On Transient.*** Positive or negative going transients may be caused by an occasional 3 ampere load current change. The amplitude of these transients will be no greater than 400 mV zero-to-peak, and will last for less than 100 ms.

(b) ***High Frequency Load-Induced Transients***. Positive or negative going high frequency transients caused by cable crosstalk, etc., will not exceed the limits shown below:

Transient Peak Amplitude (volts)	*Transient Width at 10% Amplitude Points (µsec)*
0.1	200-500
0.5	150-200
0.75	0-150

3.1.3.2.1 Power Dissipation

The power required by the instrument from the +28 Volt Main Bus will be as shown in Table 4.

3.1.3.2.2 Power Limiting

The instrument shall not limit the short circuit current drain on the spacecraft +28-Volt Main Bus. The instrument will be serviced by two 7 ampere rated fuses (one for Main Bus A and one for Main Bus B) in the spacecraft which shall not be tied together within the instrument.

3.1.3.2.3 Load Current Ripple

The peak-to-peak amplitude of steady-state load current ripple generated by the instrument shall not exceed 2 percent of the maximum average steady-state current drawn by the instrument. In the normal operating mode the fundamental frequency of load current ripple shall not be a submultiple of the frequency band 121.5 MHz ±15 kHz.

3.1.3.2.4 Transient Load Current

Exclusive of instrument turn-on, transient load currents drawn by the instrument shall not exceed 150 percent of the maximum average steady-state current drawn from the +28-volt Main Bus by the instrument, and steady-state operation shall be attained within 30 milliseconds from the start of the transient.

NOTE: The following definition of Transient and Steady-State Load Currents shall apply to this document:

Transient: Duration <50 ms, non-repetitive;
Steady-State: Duration ≥ 50 ms, non-repetitive

(1) For fuse sizing purposes, the typical worst-case and typical transient loads will be as depicted in Figures 2 and 3.

(2) The combined current drawn from all +28V Main Bus lines at instrument turn-on shall not exceed 5 amperes peak.

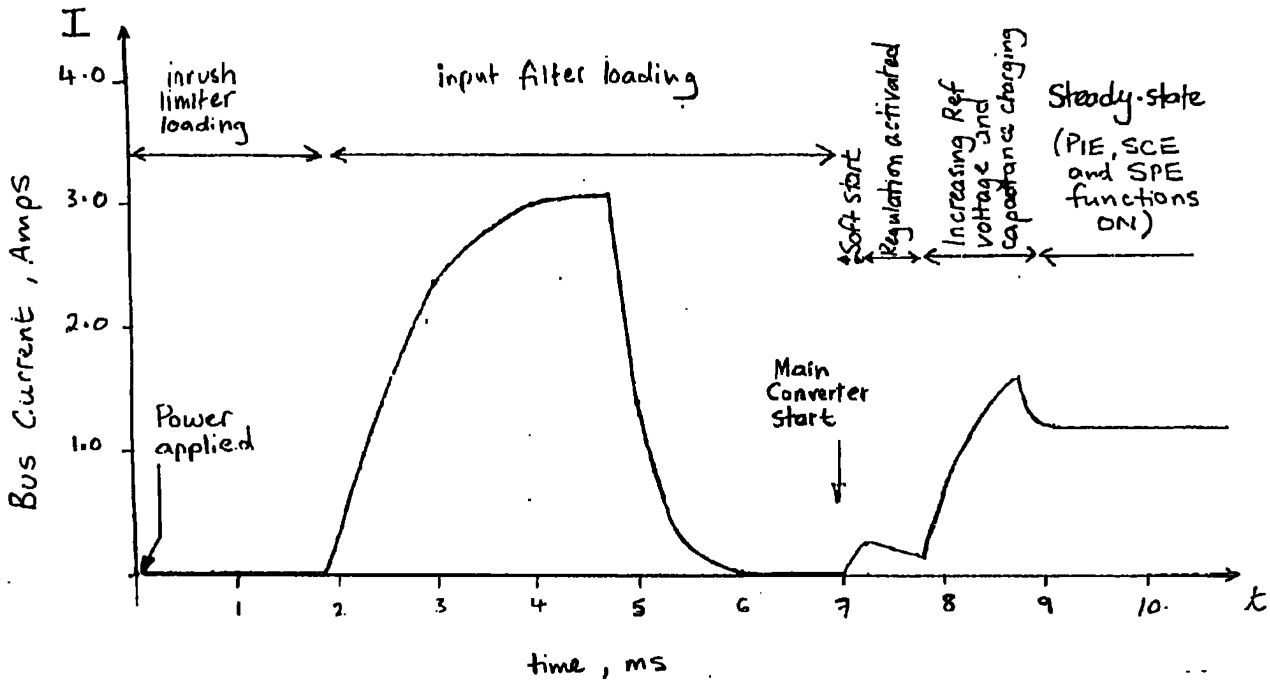


Figure 2. +28-Volt Main Bus Turn-On Transients, Typical (Sheet 1 of 2)

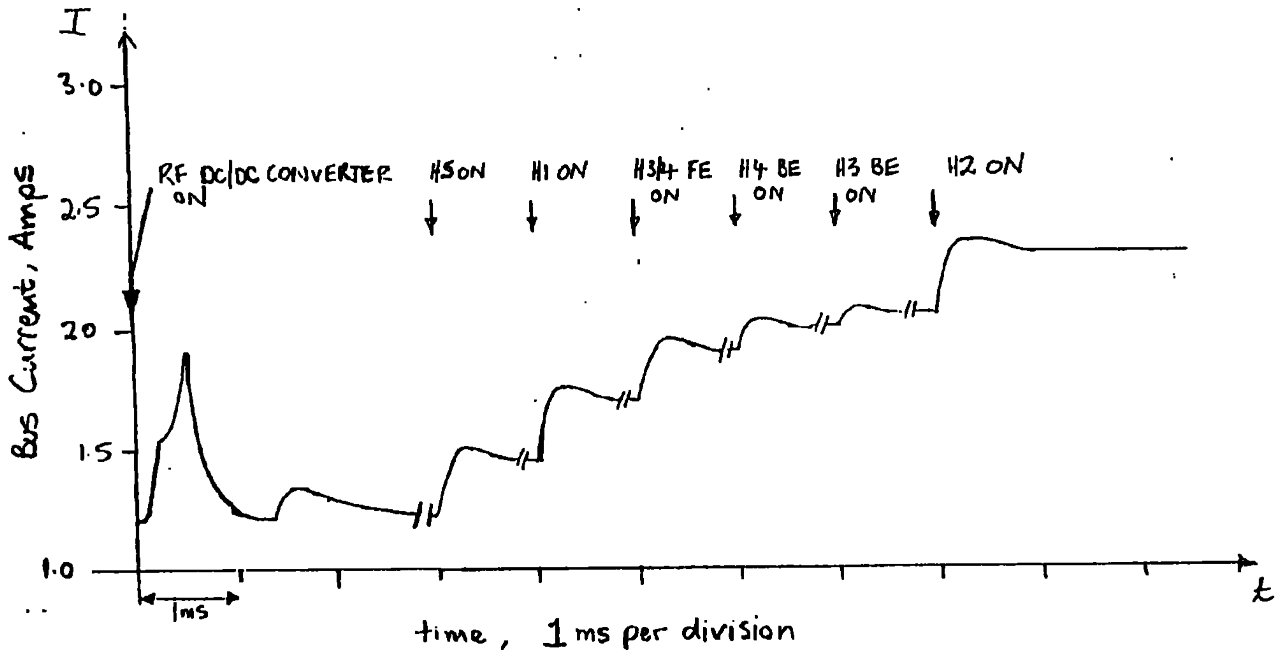


Figure 2. +28-Volt Main Bus Turn-on Transients, Typical (Sheet 2 of 2)

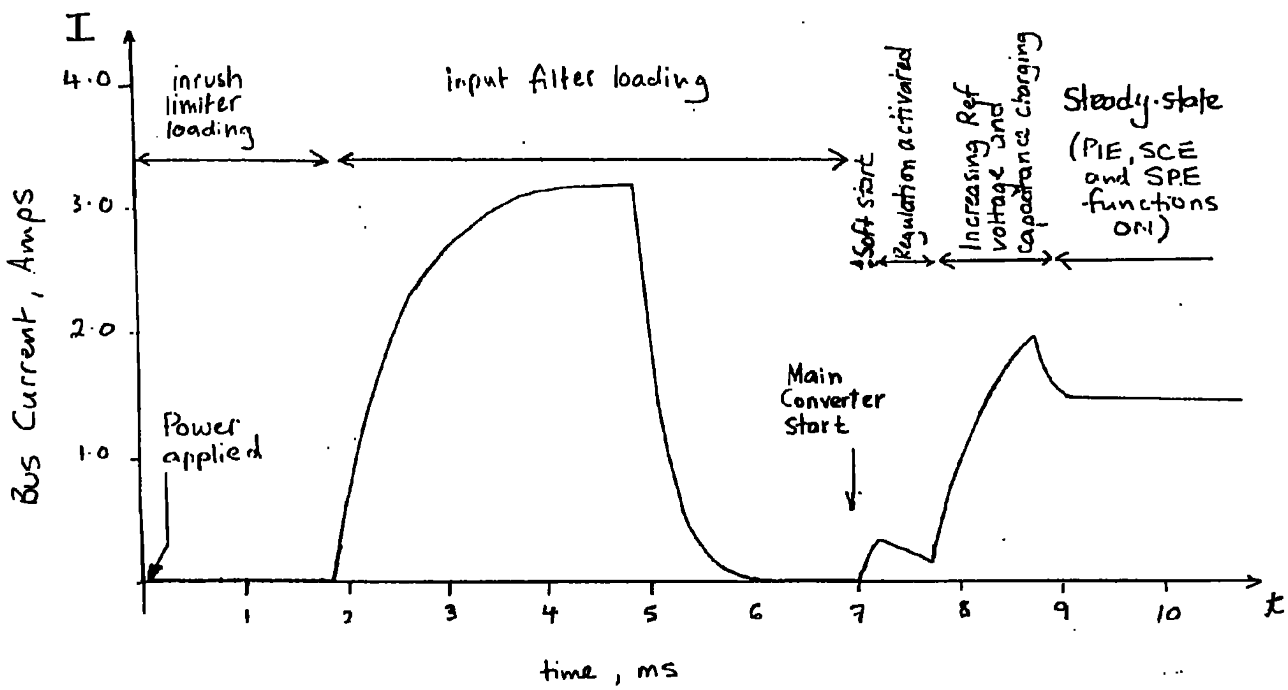


Figure 3. +28V Main Bus Worst-Case Turn-On Transients, Typical (Sheet 1 of 2)

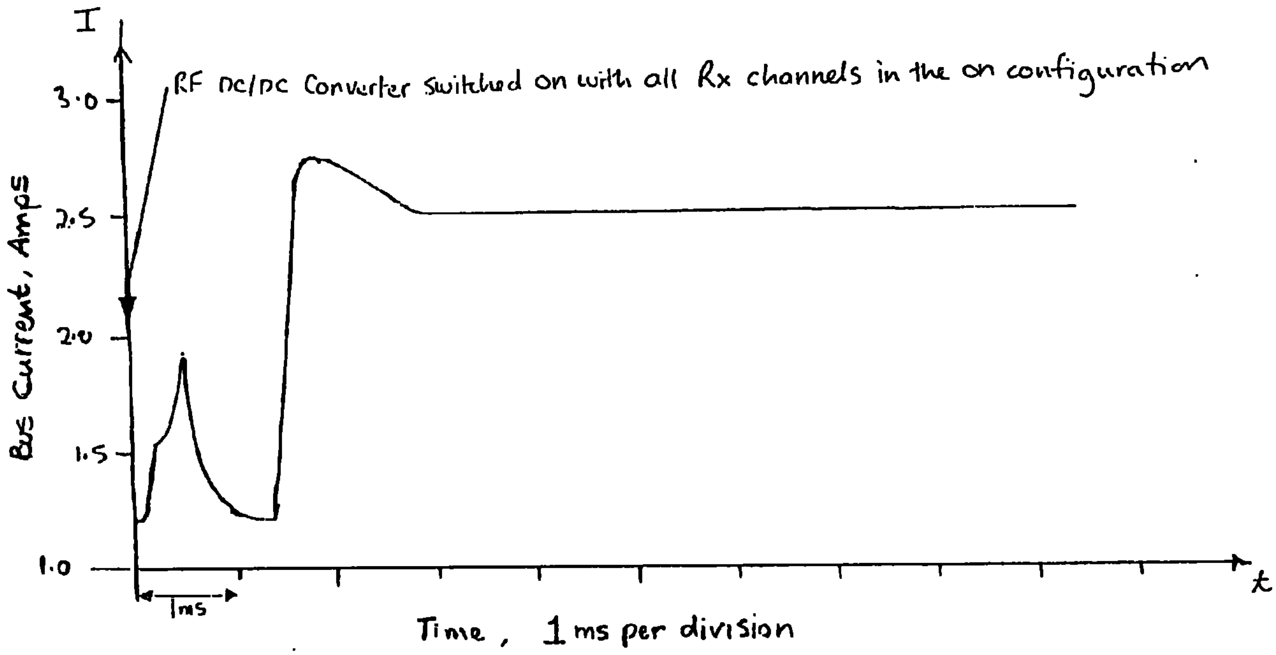


Figure 3. +28V Main Bus Worst-Case Turn-On Transients, Typical (Sheet 2 of 2)

(3) Any rate of change of current on the +28V Main Bus shall not exceed 20 mA/μsec.

3.1.3.2.5 DC/DC Converter Frequency

The DC/DC converter frequency will be 262.14 kHz except in free-running mode at initialization (2.0 sec) or if the reference clock is not present, when a variation of ±20 percent can occur.

3.1.3.3 +28-Volt Pulse Load Bus Power Requirements

The +28-Volt Pulse Load Bus is a full time regulated bus supplied to each instrument (as required) to power high current transient loads, such as stepper motors and heaters, which cannot meet the +28-Volt Main Bus current ripple and transient specifications. Separate connector pins shall be provided for the power and ground of this bus, which shall be completely isolated from other power and grounds within the instrument.

The spacecraft will provide a regulated power bus at a level of +28.0 ±0.56 volts at the instrument, exclusive of ripple and transients as specified in Items 2, 3, and 4. The PSE pulse load bus power is input to the MIU which then provides pulse load bus power and survival (pulse load) bus power to the MHS. Due to a voltage drop internal to the MIU, the pulse load bus voltage provided to the MHS will be 28.56 volts maximum and 27.01 volts minimum, and the survival (pulse load) bus voltage provided to the MHS will be 28.56 volts maximum and 25.84 volts minimum.

The voltage on the +28-volt Pulse Load Bus will not exceed +38.0 volts or drop below the +15.0 volts, including all ripple and transients.

(1) ***Source Impedance***. The small signal source impedance of the +28-volt Pulse Load Bus will not exceed 0.3 ohms at the instrument at all frequencies below 106 kHz.

(2) ***Voltage Ripple***. Sinusoidal voltage ripple on the +28-volt Pulse Load Bus will not exceed 200 millivolts peak-to-peak for frequencies up to 106 kHz, exclusive of low frequency load transients specified in Item 4.

(3) ***Source Failure Mode Transients***. The instrument shall survive, without permanent degradation, the following transients which may appear on the +28-volt Pulse Load Bus. These transients may appear when the spacecraft supply switches to a backup regulator in the event of a failure of the primary regulator. The instrument shall return to normal operation within one complete operational cycle after the bus has returned to its regulation band.

(a) ***Under-Voltage Behavior***. A voltage on the +28-volt Pulse Load Bus of less than +27 volts and more than +15 volts will constitute an under-voltage condition. This condition may exist for a period of up to 3.0 seconds.

(b) ***Over-Voltage Behavior***. A voltage on the +28-volt Pulse Load Bus of greater than 29.50 volts will constitute an over-voltage condition. Voltages greater that 30.5 volts will exist for no longer than 50 ms and will be immediately followed by an under-voltage condition for up to 1.5 seconds.

(4) ***Source Voltage Transients (Operational)***. The instrument shall continue to operate within specification when the following transients occur on the +28-volt Pulse Load Bus.

(a) ***Low Frequency Load-Induced Transients***. The output voltage transient of the Pulse Load Regulator for a step change in load current will be within the limits defined in Figure 4 over the input voltage range.

(b) ***High Frequency Load-Induced Transients***. These transients will not exceed the limits shown in Paragraph 3.1.3.2 Item 4(b).

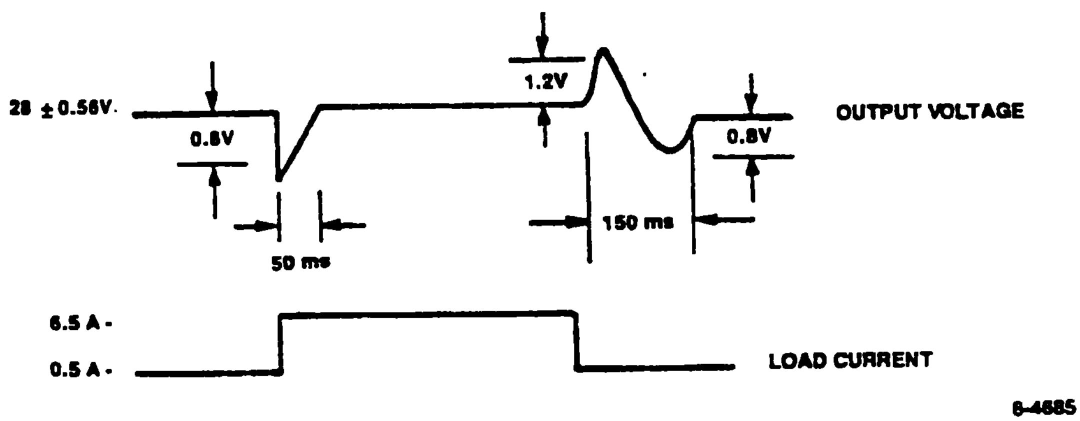


Figure 4. +28-Volt Pulse Load Bus Load-Induced Transient

3.1.3.3.1 Power Dissipation

(1) Two separate +28V Pulse Load Bus lines will be provided to the MHS: one for operational (motor, heater) power and one for survival heater (non-operational) power.

(2) The power required by the instrument from the +28-Volt Pulse Load Bus will be as given in Table 4.

(3) The typical transients on the +28-Volt Pulse Load Bus for this instrument will be as shown in Figures 5A and 5B.

3.1.3.3.2 Power Limiting

(1) The instrument will be serviced by one 5 ampere rated fuse for Pulse Load Bus (survival heater) and two 5 ampere rated fuses, one each for Pulse Load Buses A and B (scan motor and operational heaters) in the spacecraft which shall not be tied together within the instrument.

(2) The instrument shall not limit the short circuit current drain on the spacecraft +28-volt Pulse Load Bus so as to prevent blowing the spacecraft fuse in the event of a failure within the instrument.

3.1.3.3.3 Load Current Ripple

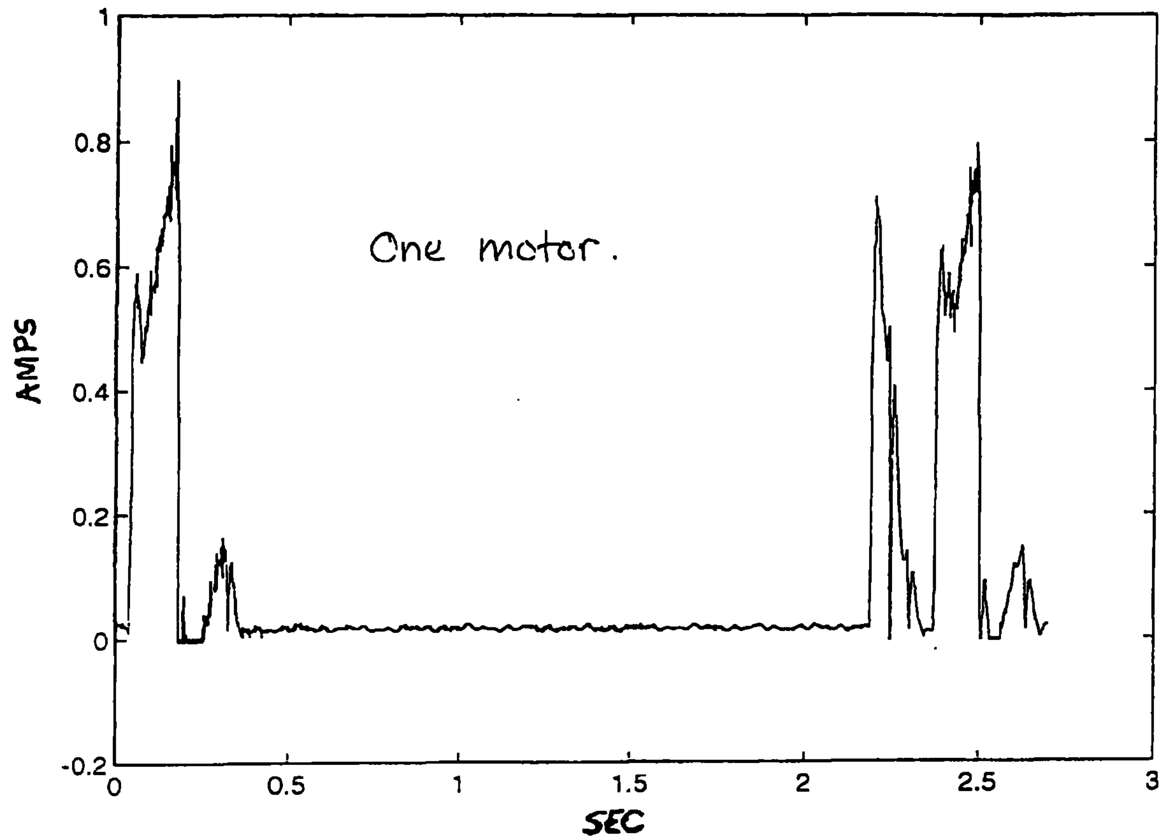
The peak-to-peak amplitude of steady-state ripple fed back by the instrument to the Pulse Load Bus shall not exceed 10 percent of the maximum average current drawn by the unit from the Pulse Load Bus. This ripple current shall be exclusive of repetitive current pulses created by stepper motor or heater switching loads. The fundamental frequency of the load current ripple shall not be a submultiple of the frequency band 121.5 MHz ±15 kHz.

3.1.3.3.4 Transient Loads

(1) Typical waveforms, including transients, for load currents drawn from the +28-volt Pulse Load Bus during different instrument operating modes will be as shown in Figures 5A and 5B. Typical turn-on transients will be as shown in Figure 6.

(2) The maximum current pulse drawn from the +28V pulse load bus shall be limited to 5.0 amperes for a period of 1.0 second maximum. The maximum instantaneous rate of change of any current pulse applied to the pulse load power circuit shall not exceed 30 mA/µsec.

(3) The rate-of-rise on the survival +28 Volt Pulse Load Bus will be 2A/µsec (maximum) at survival heater turn-on.



NOTE: Figure shows the current for one motor. Two motors are operating together, so the bus current is twice that shown (two times the motor current).

Figure 5A. +28V Pulse Load Bus Transients, Typical During Operation - Motors

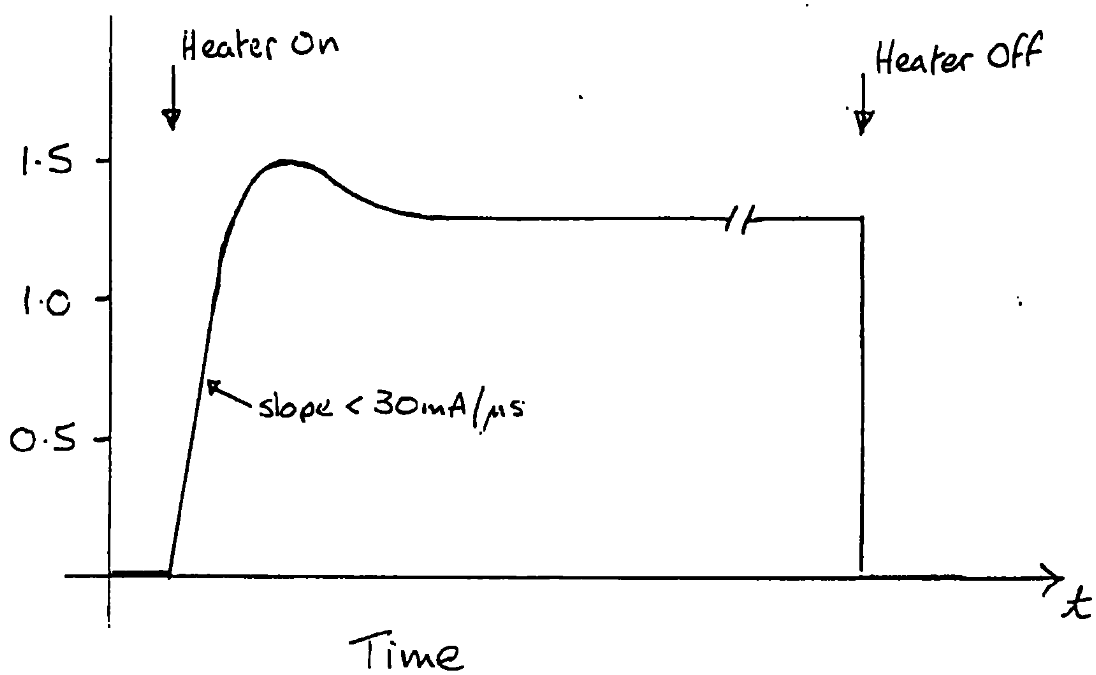


Figure 5B. +28V Pulse Load Bus Transients Typical, During Operation – Heaters On/Off

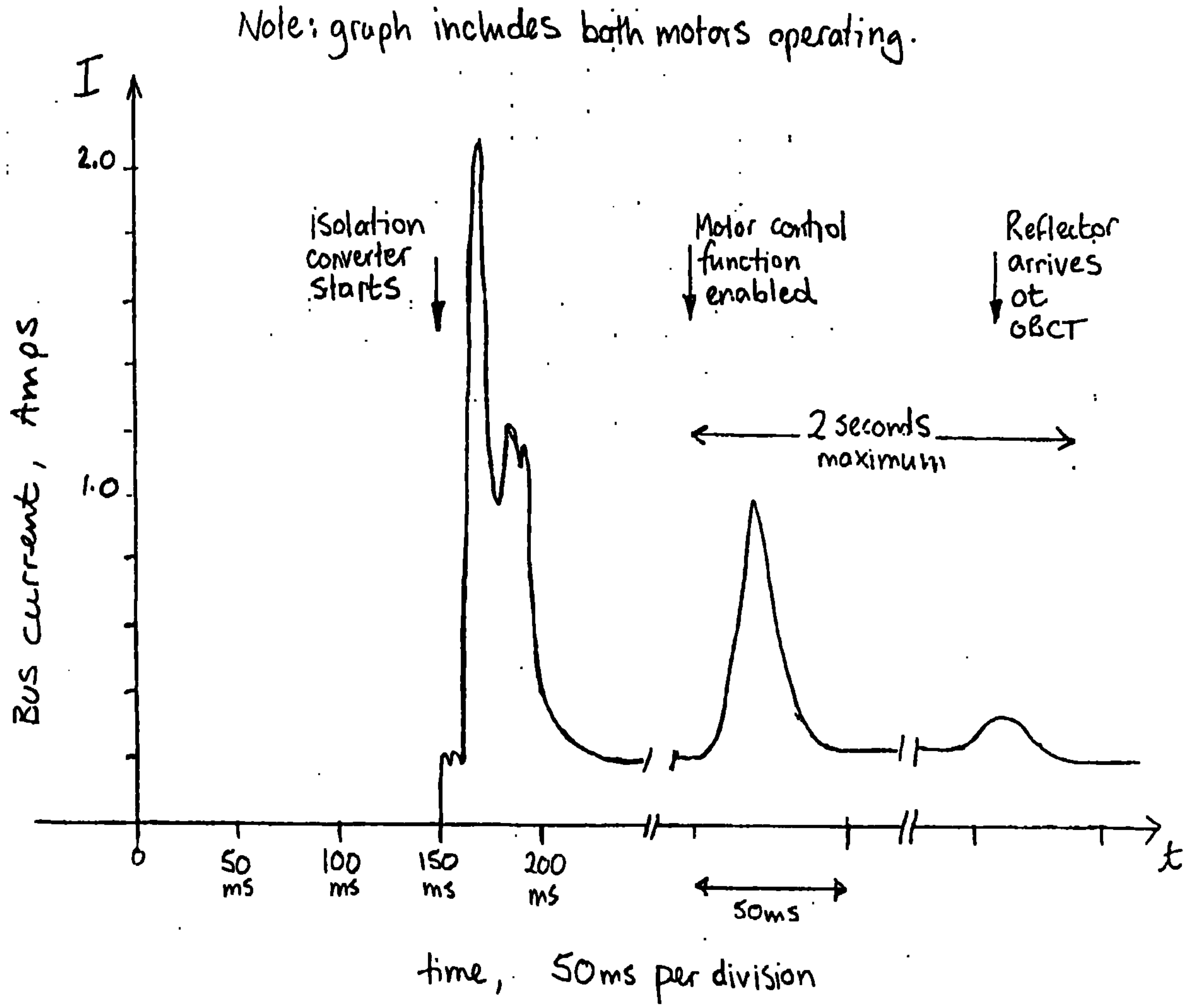


Figure 6. +28V Pulse Load Bus Turn-On Transients, Typical
(Scan Control Electronics Function Turn-On)

3.1.4 Input Timing and Control Signals

The spacecraft will provide the following input timing and control signals to the instrument. The interface circuits for the clock and the sync signal will be the Bi-Polar Clock Circuit which will be compatible with the M38510/10403. The line driver shall conform to the electrical requirements of M38510/10403, with common voltage limited to ±12V. The line receiver shall conform to the electrical requirements of M38510/10404, with common voltage limited to ±12V.

The logic ON state is represented by the '+' (or A) signal greater than the '-' (or B) signal. The logic OFF state is represented by the '-' (or B) signal greater than the '+' (or A) signal.

No spacecraft time code will be provided to the MHS instrument. The instrument shall not be damaged by the absence of the clock signal and/or the synchronization pulse signal.

3.1.4.1 Reference Clock

A clock signal shall be supplied to the instrument by the spacecraft. Characteristics of the Reference Clock signal at the input of the instrument are described below:

Signal type:	constant frequency squarewave
Frequency:	1048576 ± 10 Hertz (Coherent with the 1.248 MHz Clock)
Short term stability (1 minute):	$< \pm 5$ parts per 10^7
Frequency Drift (1 week):	$< \pm 3$ parts per 10^8
(1 year):	$< \pm 2$ parts per 10^6
Duty Cycle:	50 ± 15 percent of period
Critical timing transition:	OFF to ON logic status
Transition Time:	100 nsec maximum

The function of this clock in the instrument shall be as follows:

(1) 1.048576 MHz - Synchronization of timing of instrument functions to the spacecraft clock.

3.1.4.2 Synchronization Signal

A logic ON pulse with a repetition period of 8 seconds and a duration of 240.4 microsecond shall be supplied to the instrument. The 8 Second Sync will be derived from the spacecraft 1.248 MHz clock, not from the 1.048576 MHz clock. Its critical timing transition is ON to OFF logic status.

Characteristics of the 8 sec sync pulse at the input of the instrument are described below:

Signal Type:	Constant frequency positive pulse
Frequency:	1/8 Hertz
Short Term Stability:	<±5 parts per 10^9
Long Term Drift:	<±3 parts per 10^8 (1 week)
	<±2 parts per 10^6 (1 year)
Pulse Width Stability:	258.4 ±18 µsec

The functions of these sync signals in the instrument are as follows:

(1) 8 Second Sync - to synchronize the instrument output data format with the start of each AIP frame and to synchronize the scan operation.

3.1.4.2.1 Sync Interruption

Ground commands will be required to resume nominal operation of the instrument if the reference clock is removed. If the reference clock remains and the 8 sec sync is interrupted, then the instrument will resync itself.

When the 8 second sync pulse stops, the MHS will continue to operate nominally provided the Reference Clock is present, with the instrument timing based on the last 8 second sync pulse received. Subsequent 8 second sync pulse datums within the instrument will occur at 8 second intervals derived from the Reference Clock. If the Reference Clock is removed, the MHS will enter Fault Mode irrespective of whether the 8 second sync pulse is present or not. In this mode all science data gathering and scan control activities are stopped and the instrument needs to be commanded to restore nominal operation.

If the 8 second sync pulse is reapplied such that it is not in phase with the previous 8 second sync pulse, then the MHS will immediately start to resynchronize to the new 8 second sync pulse. However, there will be a significant disturbance to all functions, e.g., scan control, science data gathering, generation of housekeeping and science data packets. All housekeeping and science data will be invalid for one or two scan periods, but after this time the MHS will recover and resume nominal operation, with no external commanding necessary. The internal instrument software is not expected to crash.

3.1.4.3 Commands

The spacecraft will provide the command inputs listed below to the MHS. The instrument shall be able to survive commands sent in any sequence.

3.1.4.3.1 1553 Commands

All 1553 commands shall be verified through telemetry. The spacecraft level MHS command mnemonics will be as shown in Table 5. Further details on the functions of each command are given in Table 5 and in the following paragraphs. This information is defined in the MHS TM-TC and Science Data Format Directory (MHS-TN-JA063-MMP).

No more than one instrument command and one packet request shall be sent per scan period (8/3 seconds).

All operational commands will be transmitted from the spacecraft to the instrument via the 1553CCSDS Command/Telemetry Bus. This bus is defined in the MHS TM-TC and Science Data Format Directory (MHS-TN-JA063-MMP).

3.1.4.3.2 Overcurrent Protection Disable Commands

The spacecraft will provide Overcurrent Protection Disable commands to the MHS. These commands are used to disable overcurrent protection circuits of the Main Converter and the RF Converter. The electrical interface will be as defined in Figure 7.

3.1.4.3.3 Safe Commands

The spacecraft will not provide a Safe Command to the instrument. The Safe Command input to MHS will be unconnected.

3.1.5 Instrument Output Signals

3.1.5.1 General

The output data signals supplied by the instrument to the spacecraft shall be assignable into three categories – Science Data, Command/Telemetry data and Analog Telemetry.

The science data bus and command/telemetry bus are defined in the MHS TM-TC and Science Data Format Directory (MHS-TN-JA063-MMP).

3.1.5.2 Science Data

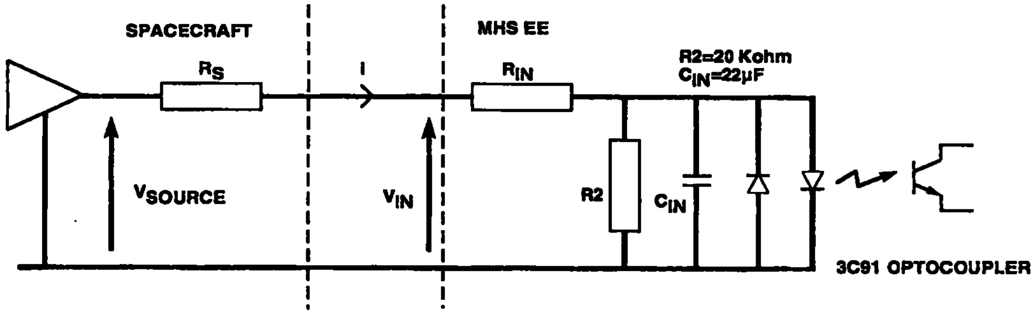
The MHS data will be in AIP minor frame words 48 through 97 as shown in Figure 8. The MHS will generate one science data packet every scan cycle (8/3 sec). The science data packet size shall be 1300 octets.

3.1.5.3 Housekeeping Telemetry

3.1.5.3.1 General

The MHS housekeeping telemetry will be inserted in TIP word 102 (AIP word 205). The MHS will generate one housekeeping telemetry packet per scan cycle (8/3 sec). The housekeeping telemetry packet size shall be 30 octets.

OVERCURRENT PROTECTION DISABLE COMMANDS



CNE14 INTERFACE SCHEMATIC

PARAMETER	MHS	COMMENTS
V_{IN} (ON)	–	2.5V IS TOO LOW TO GUARANTEE THE I/F ON
EE ANALYSIS IS DONE USING THEVENIN EQUIVALENT CIRCUIT WITH A VOLTAGE SOURCE V_{SOURCE} AND SERIES RESISTANCE R_S INSTEAD OF THE INTERFACE VOLTAGE V_{IN}		
V_{SOURCE} (ON)	4.95V MAX 3.55V MIN	RESULTS IN ABOUT 3V AT THE INTERFACE
R_S	100Ω ± 5%	
$I_{LEAKAGE}$ (SOURCE)	<10µA	100µA GIVES 2V ACROSS 20KΩ RESISTOR, WHICH TURNS THE OPTOCOUPLER ON. 10µA SHOULD BE EASY TO ACHIEVE
C_{IN}	22µF ± 40%	22µF IS NOT SIGNIFICANT FOR SOURCE AS THE SERIES RESISTOR LIMITS THE INRUSH
IMAX	10mA	10mA MAX IS DERATED CURRENT FOR OPTOCOUPLER
OVERVOLTAGE	6V (AT V_{SOURCE})	10V GIVES 20mA IN OPTOCOUPLER (IE. 2 X DERATED VALUE) AND MAX RATED POWER DISSIPATION FOR R_{IN}
R_{IN}	357Ω ± 3.6%	

MMS-P REQUIRE FROM LMAS:
- V_S MIN, MAX VALUES
- R_S MIN, MAX VALUES
- I LEAKAGE VALUE, MAX

97-0297M

Figure 7. Overcurrent Protection Command Interface Diagram

3.1.5.3.2 Housekeeping Telemetry Points

The housekeeping telemetry points shall be as defined in the MHS TM-TC and Science Data Format Directory (MHS-TN-JAO63-MMP).

3.1.5.4 Analog Telemetry

3.1.5.4.1 General

A 16-second subcom, generated by the TIP, will be used to sample the three MHS survival analog telemetry points.

TABLE 5. SPACECRAFT/MHS COMMAND INTERFACES

(deleted)

Refer to MHS-TN-JA063-MMP

(Applicable Document 2.1.2 (14))

3.1.5.4.2 Analog Telemetry Points

The Analog Telemetry Points used by the MHS will be shown in Table 6.

The MHS will be provided with three Analog Telemetry channels to monitor the survival thermistors when the instrument power is off. The minimum and maximum thermistor resistance shall be 1.6K ohms (min) and 100 Kohms (max).

3.1.5.5 Main Bus Selection Switch Status

The MHS will provide a single telemetry bit indicating which main bus has been selected. A logic 1 will indicate Side A. A logic 0 will indicate Side B.

3.1.6 Test Points

The test points detailed below will be used as required by the instrument contractor during test of the MHS. These points will not be used by the spacecraft and will not be included in the spacecraft harness. The instrument contractor shall supply flight covers for any Test Connectors.

3.2 Mechanical Interface

3.2.1 Physical Characteristics

3.2.1.1 Dimensions

The MHS shall consist of one unit; the outside dimensions of which, including the mounting feet, shall not exceed the dimensions as shown in Figure 9A. The mounting hole pattern for the MHS shall be as shown in Figure 9B.

The following interface data shall be indicated in the instrument configuration drawing:

(1) Mounting hole location and tolerance
(2) Connector location and keying
(3) Center of gravity location
(4) Mass Properties: Inertia - X, Y, and Z axes and mass
(5) Sunshield location (if one is used)
(6) Harness tie points (if required)
(7) Identification marking
(8) Ground Strap Stud (attach point for spacecraft contractor-provided ground strap)
(9) Location of the optical cubes
(10) Scan plane and origin of FOV.

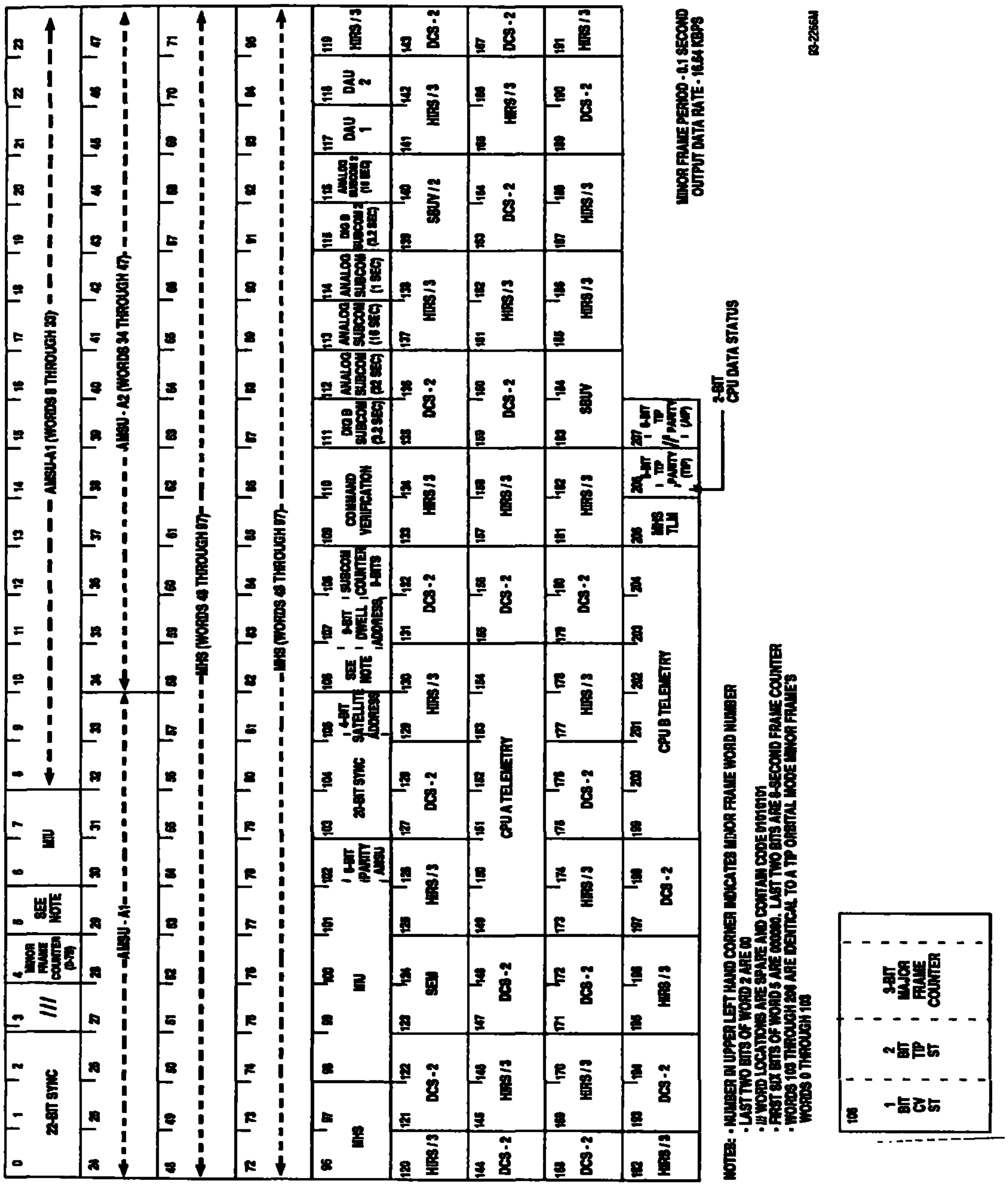


Figure 8. AIP Minor Frame Format

TABLE 6. MHS ANALOG TELEMETRY

No.	Telemetry Point Name	Range	Scale Factor	CH# (subcom)
1	Receiver Temperature	-60 to +55	**	16AN17 333 (16-1)
2	Electronics Equipment Temperature	-60 to +55	**	16AN16 317 (16-1)
3	Scan Mechanism Temperature	-60 to +55	**	16AN80 365 (16-1)
Notes: These telemetry points refer to three Type GB32 survival thermistors which the instrument provides as discrete devices. These telemetry points are powered even when the instrument power is off. **To be provided in the MHS Spacecraft Level Test Manual for each Flight Unit.				

3.2.1.2 Weight

The total weight of the instrument shall not exceed 145.2 pounds (66 kg). It shall be determined to an accuracy of one percent of the unit weight.

This total weight shall include instrument contractor-supplied flight items (e.g., thermal blankets).

3.2.1.3 Moments of Inertia

The total moments of inertia of the instrument (about the center of gravity) are as follows:
$I(V,Y) = 2.70$ kg-m^2
$I(A,Z) = 3.81$ kg-m^2
$I(N,X) = 4.89$ kg-m^2

3.2.1.4 Disturbance Torque

The MHS disturbance torque limits are as shown in Figure 9C.

3.2.1.5 Center of Mass

The instrument center of mass locations shall be established to an accuracy of 5 mm or better in each direction.

The maximum distance from the spacecraft panel to the instrument center of mass of the MHS shall be no greater than 215 mm. The location of the center of mass shall be shown on the MHS Instrument Configuration Drawing No. MMS 3175-FA002-25-0.

3.2.2 Instrument Mounting

3.2.2.1 Instrument Mounting Surface

The instrument mounting flange surfaces shall be within the envelope defined in Para. 3.2.1. The mounting surface flatness shall be less than 0.1 mm per 100 mm.

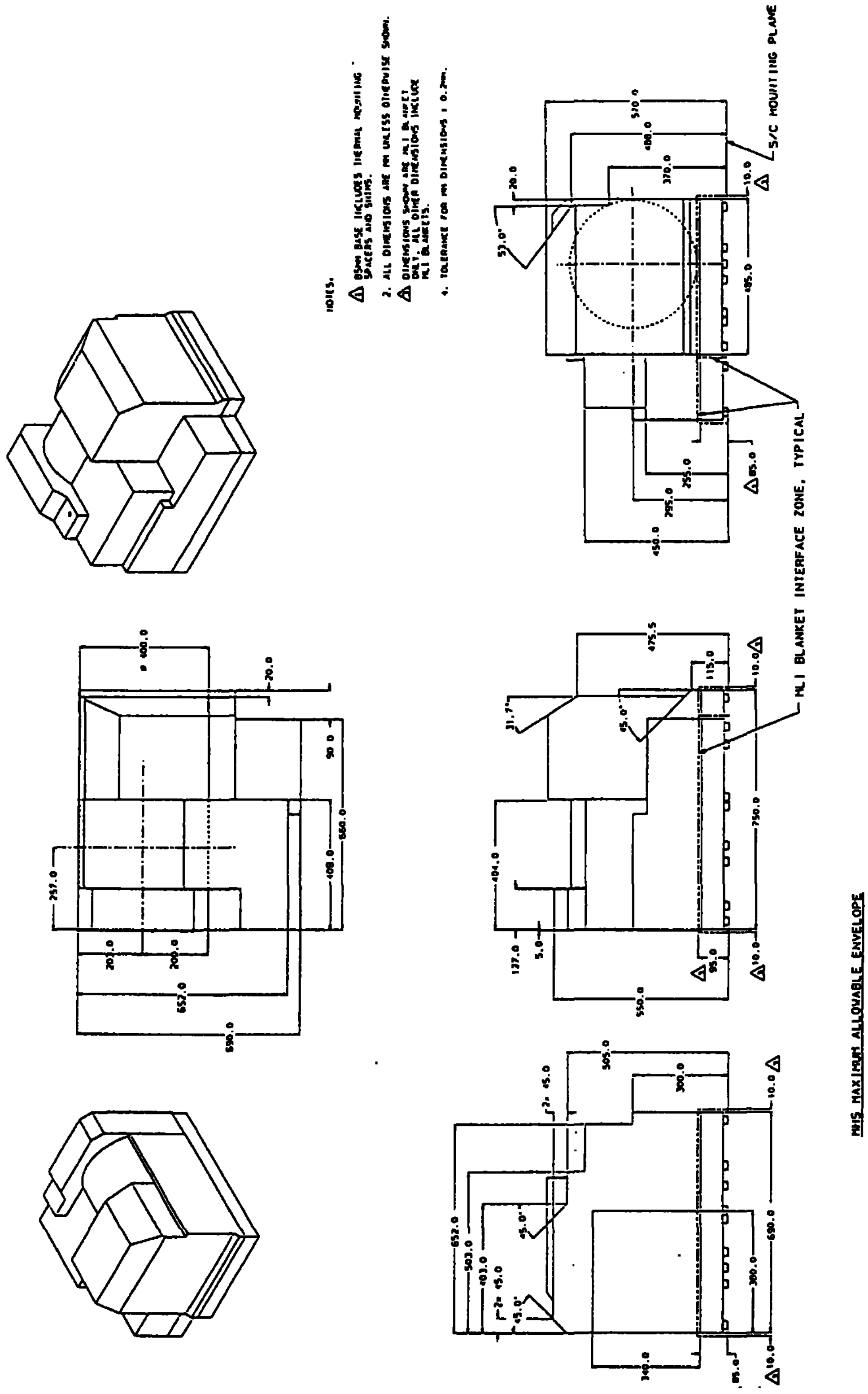


Figure 9A. MHS Outline Drawing

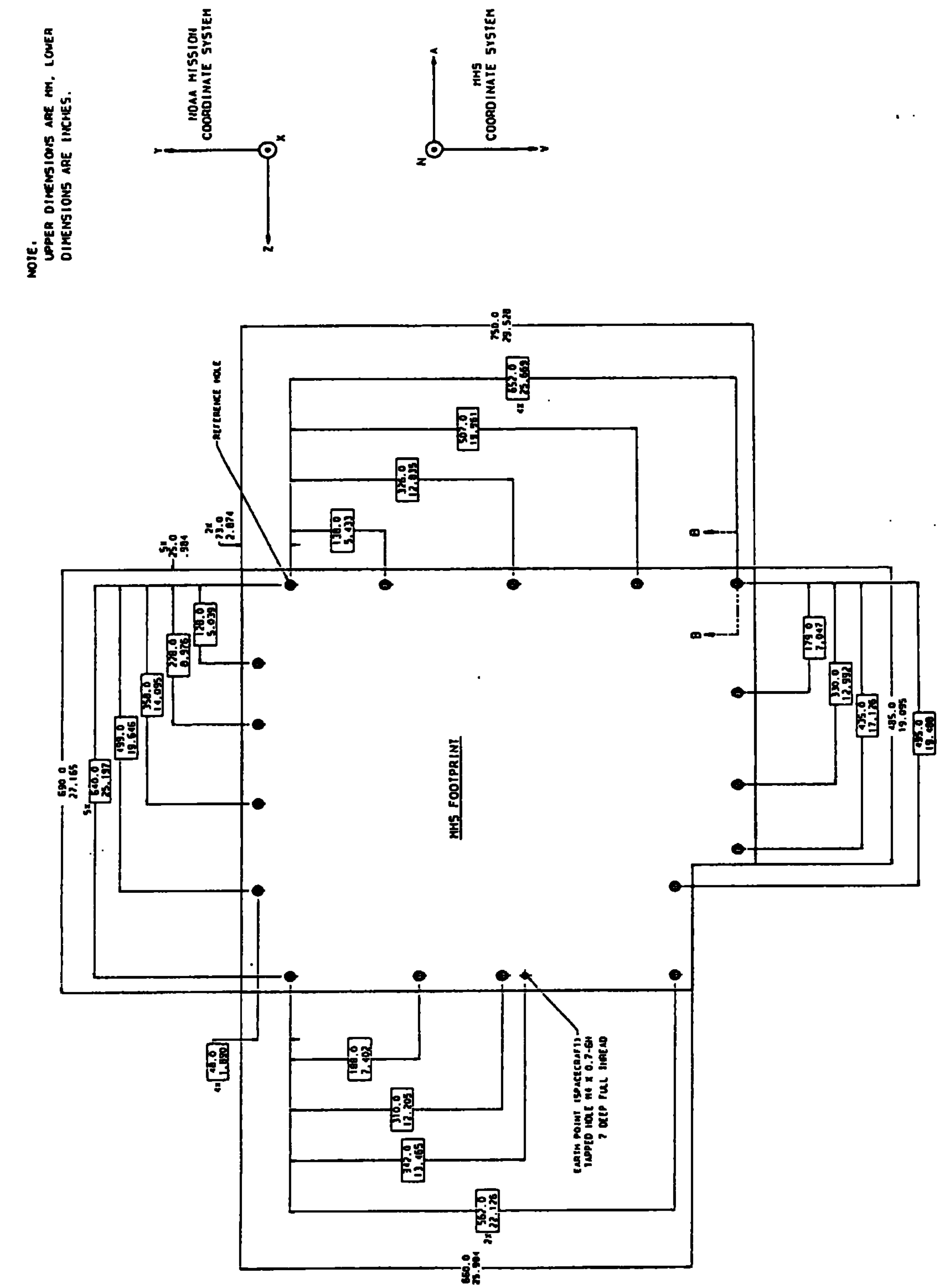


Figure 9B. MHS Mounting Hole Locations

3.2.2.2 Mounting Hole Position

The mounting hole positions shall be defined by the drill jig specified below:

Jig	*Drawing No.*
Spacecraft Pattern	3175-35025-29-1

This jig shall have a reference edge parallel to the spacecraft Y-Y axes.

3.2.2.3 Instrument Location

The MHS location and orientation on the spacecraft will be in accordance with the following spacecraft drawings: ATN spacecraft assembly (Martin Marietta Dwg. 20041556), ATN-ESM assembly (Martin Marietta Dwg. 3278775), and TIROS N & N'Spacecraft Field of View (Martin Marietta Dwg. 20041663).

3.2.2.3.1 Accessibility

Instruments shall be designed for installation and removal from the spacecraft without disassembly of the instrument. Instruments will be mounted to the spacecraft by means of mounting bolts passing through flanges located on the instrument. All mounting bolts shall be accessible from the top (instrument side) of the spacecraft structure. Accordingly, no threaded or "blind" mounting holes shall exist in the instrument. Access restrictions to the MHS bolts will be identified in the instrument ICD and special tooling will be provided as necessary.

3.2.2.4 Spacecraft Mounting Surface

The spacecraft ESM consists of aluminum honeycomb with threaded inserts potted into the honeycomb at the appropriate instrument mounting locations. The planarity of the honeycomb surface will be 0.010 inch/foot maximum.

The instrument will be shimmed, if necessary, to ensure proper alignment with respect to the AMSU-A1 reference. Instrument shimming is the responsibility of the spacecraft contractor.

This instrument will utilize 1/4 inch diameter hardware for mounting purposes. The MHS will be mounted externally to the ESM earth-facing panel through inserts. The pullout strength of these are:

TABLE 7. PULLOUT STRENGTH AND SHEAR FORCE OF INSERT IN ESM EARTH-FACING PANEL

Hardware	Pullout Strength		Shear Force	
	(kg)	(pounds)	(kg)	(pounds)
Insert Mount	341	750	909	2000

For each instrument-mounted assembly, disturbance torque, force and momentum shall be determined and limited in accordance with the following calculations[1]:

- For disturbances in the frequency range $0 < \omega < 0.062$ rad/sec (the control bandwidth), let $S_1(\omega)$ be the power spectrum of the disturbance momentum vector $(h_v, h_a, h_n)^T$, and

$$G_1(\omega) = \begin{bmatrix} 0 & 0 & 2.4(10^{-3}) \\ 0 & 3.2(10^{-3})j\omega & 0 \\ 2.4(10^{-3}) & 0 & 0 \end{bmatrix}, \quad j = \sqrt{-1} \; .$$

- For disturbances in the frequency range $0.062 \le \omega < 0.8$ rad/sec (rigid body regime outside the control bandwidth), let $S_2(\omega)$ be the power spectrum of the disturbance torque/force vector $(\tau_v, \tau_a, \tau_n, f_v, f_a, f_n)^T$, and

$$G_2(\omega) = \frac{1}{\omega^2}\begin{bmatrix} 1.6(10^{-4}) & 0 & 0 & 0 & 8.7(10^{-5}) & 4.0(10^{-5}) \\ 0 & 1.5(10^{-4}) & 0 & 8.5(10^{-5}) & 0 & 2.0(10^{-4}) \\ 0 & 0 & 8.8(10^{-5}) & 2.2(10^{-5}) & 1.1(10^{-4}) & 0 \end{bmatrix} .$$

- For disturbances in the frequency range $0.8 \le \omega < \infty$ rad/sec (the regime of flexible-body modes), let $S_3(\omega)$ be the power spectrum of the disturbance torque/force vector $(\tau_v, \tau_a, \tau_n, f_v, f_a, f_n)^T$, and

$$G_3(\omega) = \frac{1}{\omega^2}\begin{bmatrix} 2.2(10^{-2}) & 5.6(10^{-5}) & 6.7(10^{-4}) & 1.7(10^{-4}) & 2.0(10^{-3}) & 8.5(10^{-3}) \\ 6.7(10^{-5}) & 3.1(10^{-4}) & 7.7(10^{-4}) & 4.9(10^{-4}) & 5.7(10^{-6}) & 1.6(10^{-5}) \\ 4.5(10^{-4}) & 7.7(10^{-4}) & 3.1(10^{-3}) & 2.0(10^{-3}) & 5.7(10^{-5}) & 2.0(10^{-4}) \end{bmatrix} .$$

- Compute the following summation over all discrete frequencies ω_k at which spectral lines exist, including negative frequencies since $S(\omega_k) = S(-\omega_k)$:

$$C_i = \sum_{k=-\infty}^{\infty} G_i(\omega_k) S_i(\omega_k) G_i^T(-\omega_k) , \quad i = 1,2,3 \; .$$

To ensure compliance with spacecraft pointing and jitter requirements, all of the following conditions shall be met:

1. The total spacecraft pointing response to disturbances shall be within the following limit:

$$\begin{pmatrix} \sigma_v^2 \\ \sigma_a^2 \\ \sigma_n^2 \end{pmatrix} = diag\ (C_1 + C_2 + C_3) < \begin{pmatrix} 5.0(10^{-10}) \\ 5.0(10^{-10}) \\ 5.0(10^{-10}) \end{pmatrix} radians^2$$

2. Momentum about the pitch axis at any frequency shall be less than 0.5 Nms.
3. Bias momentum about roll and yaw shall be limited to 0.5 Nms.

---

[1]All units are metric - N, Nm, Nms, etc. - and radians. The roll, pitch, and yaw axes are denoted respectively, v (velocity), a (anti-sun), and n (nadir). Elements of G_1 have units of rad/(N m s) and elements of G_2 and G_3 have units rad/(kg·m^2) in the left three columns and units rad/(kg·m) in the right three columns. The 3 X 3 matrix S_1 has units (Nms)^2. The 6 X 6 matrices S_2 and S_3 have units (Nm)^2 for the first three diagonal elements and units N^2 for the last three diagonal elements. In general, the power spectrum matrices S_1, S_2, and S_3 are usually diagonal, i.e., the cross power spectra are usually assumed to be zero.

Figure 9C. Disturbance Torque Profile Limits for MHS

3.2.3 Mechanisms

3.2.3.1 Caging

None required.

3.2.4 Fields-of-View

The required MHS fields-of-view are as follows:

	Parameter	*Value*
(1)	Beam Free Projected Aperture	240 mm (includes necessary margin with respect to reflector physical aperture)
(2)	Beam Divergence (half angle)	<3 degrees
(3)	Beam Axis Scan Range	+50 to -81 degrees from +X (Nadir)
(4)	Scan Plane	Crosstrack

Note: Beam divergence shall be added to the beam axis scan in any direction.

3.2.4.1 Instrument Requirements

The clear fields-of-view for the instrument provided by the spacecraft will be defined as shown in Figure 9D. The "Earth", "Nadir", or "+X" direction shall be the "0" degree reference. The spacecraft velocity vector is in the "-Y" direction. Instrument N (Nadir) is spacecraft "+X"; V (Velocity) is spacecraft "-Y"; and A (anti-sun) is spacecraft "-Z."

3.2.4.2 Spacecraft Provisions

The spacecraft will provide the following unobstructed fields-of-view:

-36.45 to -174 degrees from +Z

3.2.5 Alignment

The in-orbit uncertainties are shown in Table 8A and include uncertainties due to launch, gravity and thermal gradients. The determination of these uncertainties for the AMSU-A and MHS instruments are based on previous analysis. It is expected that the possible movement of the AMSU-A1 and A2 modules due to vibration or launch will be less than shown due to the utilization of shear pins between the module base and the ESM.

To provide for the best possible co-registration between the AMSU-A modules and the MHS, repositioning of the instruments will be required during the initial alignment sequence. The AMSU-A2 module will utilize Martin Marietta supplied interface plates when mounted to the spacecraft. These

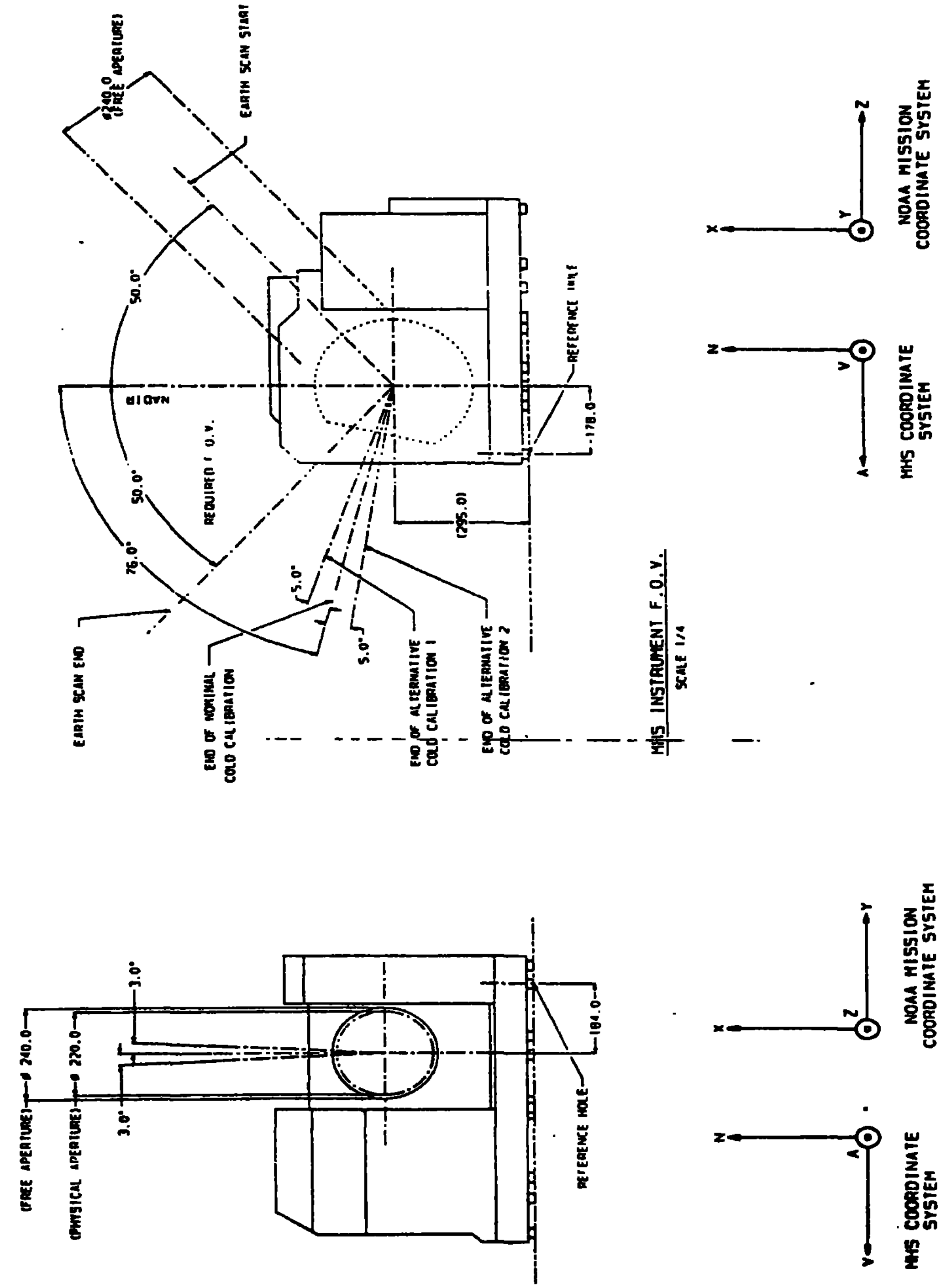


Figure 9D. MHS Fields of View

plates have been designed to have oversized mounting holes which are required to allow for maximum adjustment (rotation) about the X-axis. The mounting bolt through-holes in the MHS baseplate allow for rotation about the X-axis to meet the placement requirement. Adjustment of the AMSU-A2 and MHS about the Y- and Z-axes will be accomplished via shimming. The AMSU-A1 module, as designed, does not allow for maximum (rotation) about the Z-axis. Adjustment of the AMSU-A1 module about the X- and Y-axes will be accomplished via shimming.

To utilize the available adjustments in the AMSU-A1, AMSU-A2, and MHS and provide for the best co-registration between instruments, the following alignment scenario will be performed:

The AMSU-A1 module will be aligned relative to the primary reference axis (as defined by the ESA) with an initial placement of 0.05 degrees or less in the X- and Y-axes. AMSU-A1 will be placed, accepting the Z-axis position. AMSU-A2 will then be aligned relative to the AMSU-A1 module with an initial placement of 0.05 degrees or less in all three axes. The MHS will then be aligned relative to the AMSU-A1 module with an initial placement requirement of 0.05 degrees or less in all three axes. The worst case on ground Co-Registration between the AMSU-A and MHS instruments and the ESA will be as shown in Table 8B. Using this method the AMSU-A and MHS instruments will be able to meet the in-orbit alignment requirements relative to the primary reference axis as shown in Table 8C with an in-orbit Co-Registration between the AMSU-A and MHS instruments and the ESA as shown in Table 8D.

3.2.5.1 Alignment of the Electrical Boresight

The relative alignment of the MHS baseplate and the electrical boresight with respect to the optical reference cube will be known to an accuracy better than ±0.01 degrees. The optical reference cube will be permanently attached to the instrument and will not interfere with the instrument operation.

The nadir (+X axis) will be normal to the mounting plane (ABCD) for all instruments having alignment requirements. (See Figure 10.)

3.2.5.2 Alignment Mirrors

The MHS will provide an alignment cube.

The normal to one alignment reference mirror shall point in the Earth (nadir) direction. That mirror shall be visible along its normal when the instrument is mounted on the spacecraft.

The normal to the second alignment mirror shall be placed on a perpendicular axis such that the mirror can be seen (bore-sighted) when the unit is mounted on the spacecraft.

The shift of the electrical boresight when exposed to the test and launch environments shall not be greater than that specified in Table 8A.

TABLE 8A. AMSU-A/MHS ON-ORBIT UNCERTAINTIES (WRT ESA)

INST.	A	B	C	D	E	F	G	H	I	RSS
A1	0.1	0	0.05	0.0014	0.0014	0.028	0.05	0.025	0.0014	0.1281
A2	0.1	0	0.05	0.0014	0.0014	0.028	0.05	0.083	0.0014	0.1506
MHS	0.12	0	0.00	0.0014	0.0014	0.028	0.05	0.083	0.0014	0.1568

INSTRUMENT

A: Knowledge of Inst Mirror WRT Optical Axis

B: Repeatability of Inst Mirror Placement

C: Change in Inst Optical Axis Due to ENV Test

SPACECRAFT

D: Measurement Tolerance from RPPA to ESA Mirror

E: Repeatability of ESA Mirror

F: Uncompensated Gravitational Tolerance Between Inst

G: Change in Position Due to Launch

H: Change in Position to On Orbit Thermal Gradients

I: Measurement Tolerance from RPPA To Inst. Mirror

TABLE 8B. AMSU-A/MHS INITIAL GROUND CO-REGISTRATION

Instrument	Initial Placement (Max)	Initial Placement (RSS'D)
ESA/AMSU-A1	0.05°	0.05°
ESA/AMSU-A2	0.10°	0.07°
ESA/MHS	0.10°	0.07°
AMSU-A1/AMSU-A2	0.05°	0.05°
AMSU-A1/MHS	0.05°	0.05°
AMSU-A2/MHS	0.10°	0.07°

TABLE 8C. NEW AMSU-A/MHS ALIGNMENT REQUIREMENTS (WRT ESA)

Inst.	Initial Position (1)	Max Alignment Shift Due To Vib (2)	Final Position	Uncertainty (3)	Calculated Requirement
AMSU-A1	<0.05°	±0.05°	< 0.1°	±0.13°	0.23°
AMSU-A2	<0.10°	±0.05°	< 0.15°	±0.15°	0.30°
MHS	<0.10°	±0.05°	< 0.15°	±0.16°	0.31°

1) Sensor Optical Axes WRT S/C Primary Axis
2) Sensor Reference Axes WRT S/C Primary
3) From RSS'D Uncertainty in Table 8A

To be updated for MHS

TABLE 8D. AMSU-A/MHS IN-ORBIT CO-REGISTRATION

Instrument	Final Position	Uncertainty	In-Orbit Co-Registration
ESA/AMSU-A1	0.10°	0.13°	0.23°
ESA/AMSU-A2	0.15°	0.15°	0.30°
ESA/MHS	0.15°	0.16°	0.31°
AMSU-A1/AMSU-A2	0.10°	0.20°	0.30°
AMSU-A1/MHS	0.10°	0.21°	0.31°
AMSU-A2/MHS	0.15°	0.22°	0.37°

3.2.5.3 Alignment Measurement

3.2.5.3.1 Electrical Boresight to Mounting Hole Pattern

The instrument contractor shall measure the linear position of the electrical boresight with respect to the reference hole to verify that the instrument design is compatible with Section 3.2.5 of this document. The precision of this measurement shall be 0.05 mm.

3.2.5.3.2 Electrical Boresight to Alignment Mirrors

The instrument contractor shall measure the electrical boresight with respect to the alignment mirrors.

3.2.5.4 Reference Surfaces

The instrument and alignment reference surfaces shall be as follows:

(1) Size: 25 mm cube
(2) Perpendicularity between faces: 15 arc sec

Flight covers shall be provided for all external alignment cubes.

3.2.6 Protective Covers

Protective covers for the antenna will be required for the MHS as follows:

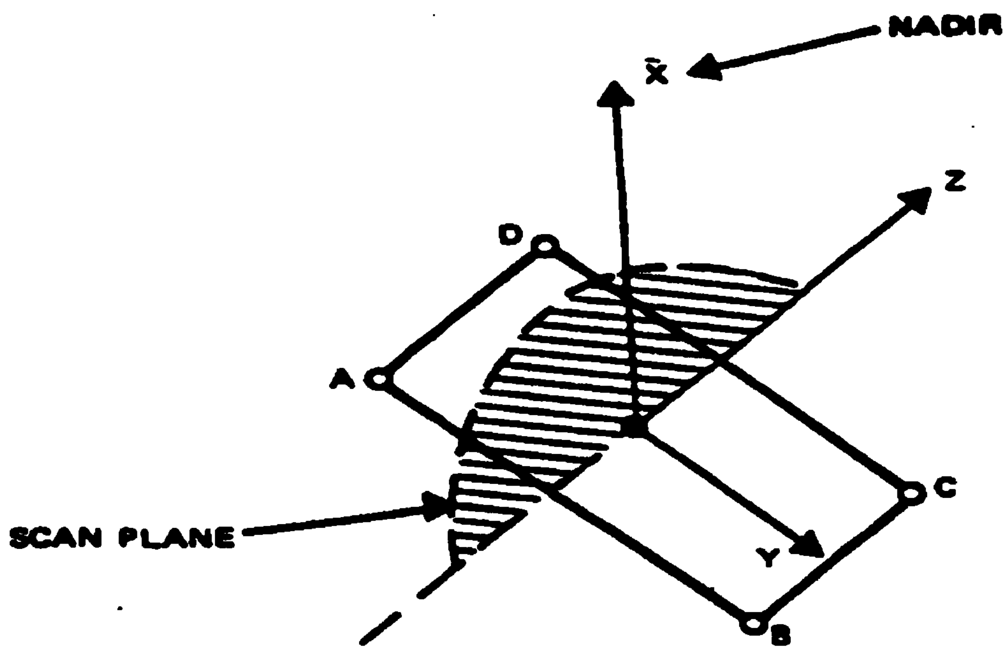
A protective cover will be installed over the MHS reflector and shroud. This cover will be designed such that the reflector can rotate with the cover in place.

Protective covers, including provision for desiccant if necessary, shall be provided by the instrument contractor to preclude entrance of foreign particles to sensitive areas and to preclude damage during handling, storage on or off the spacecraft, assembly, integration, bench test, and spacecraft test. Covers for all connectors shall be provided.

Covers required during spacecraft ground operations shall be fastened in a manner suitable to prevent loss during transportation and handling of the spacecraft with any axis down. All such items to be removed prior to flight shall be capable of removal without disassembly of any portion of the spacecraft or instrument. Such items shall be color coded in a red finish and labeled accordingly.

3.2.6.1 Removal Requirements and Reasons

Protective covers shall be removed at the launch site prior to encapsulation into the launch vehicle fairing. This will be the responsibility of the spacecraft contractor.



XZ = SCAN PLANE
ABCD = MOUNTING HOLE SURFACE
Y NORMAL TO SCAN PLANE
Y' SHALL BE PARALLEL TO SURFACE ABCD
Y SHALL BE PERPENDICULAR TO SIDE AB OR BC.

Figure 10. Optical Axis Alignment

3.2.6.2 Precautions

The handling precautions shall be:

Reference: MHS Instrument Integration Constraints Document, MHS-TN-JA271-MMP

3.2.7 Spacecraft Harness Clamp Requirements

There is no requirement for the instrument contractor to install a spacecraft Harness Clamp to the instrument. If a harness tie-down clamp is required, it will be potted to the instrument, at a mutually acceptable location, at the time of integration of the instrument with the spacecraft.

3.2.8 Marking

All equipment shall be marked with the information listed below. Marking shall be as permanent as the normal life expectancy of this article. Markings shall be resistant to chipping and located away from points of physical wear.

Manufacturer's Name or Trademark
Manufacturer's Part Number
Manufacturer's Serial Number

Marking shall not affect the leakage path between conductors or any other factor of performance.

All labels shall be easily seen and not hidden by units and parts. All connectors, test points, and adjustments shall be clearly labeled.

3.3 Thermal Interface

The basic characteristics of the instrument/spacecraft (ESM) interface and the requirements necessary to establish and maintain this interface shall be as follows.

3.3.1 Responsibility

3.3.1.1 Instrument Vendor

The instrument vendor shall be responsible for the thermal design of the instrument.

The instrument vendor shall furnish to the spacecraft contractor a complete documentation package clearly defining the physical outline of the instrument, its multilayered insulation blankets, fixed-area radiators, its louver/radiator assemblies (if any) and its mounting scheme. This documentation shall consist of a set of fully annotated drawings.

The instrument vendor shall furnish to the spacecraft contractor a reduced thermal model of the instrument for the purposes of performing systems type thermal analyses.

Reduced Thermal Models (RTMs) shall be provided for each of three (3) cases ($\gamma = 0°$, $27.5°$ and $80°$). The RTMs shall be presented in standard SINDA format or, if this is not possible or feasible, in the tabular form described below. Each of the RTMs (whether in SINDA format or in tabular form) shall be provided as a single file included on a 3.5" diskette. A printed listing of each file shall also be provided. In addition to this listing, MMS shall provide a listing of the input file (s) to their thermal analysis program and a listing of the output obtained from that input.

Diagrams/sketches describing and elucidating the RTMs shall be provided as required.

The Reduced Thermal Models of the MHS shall be limited to no more than 35 nodes each exclusive of any boundary or ancillary nodes which constitute either item's thermal environment. Thermal couplings to these external nodes shall be included in the RTMs.

The external layer of any Multi-Layered Insulation Blanket (MLIB) shall be included as a node in the RTM; an MLIB shall not be represented by substituting effective values for the thermal surface properties of the surface (node) covered by that MLIB.

The units of measurement preferred by Astro Space are: Length, inches; Mass, pound mass (lbm); Time, seconds; Power, watts; Energy, joules; and, Temperature, °C or kelvins. Other units, however, shall be acceptable. The units used shall be clearly stated. Radiation coupling factors (coefficients) shall be specified as areas (e.g., sq. inches) - not as areas multiplied by the Stefan-Boltzmann constant.

Time-dependent parameters may be defined by an explicit formula or by an array of parameter values and an associated time-array. The start time (=0) for all time-dependent parameters shall be at that point in orbit when the spacecraft is in orbital conjunction with the sun. For orbits which are eclipsed by the earth's shadow, this point is frequently referred to as "orbital noon" or "orbital noontime".

MMS shall provide descriptions of all active thermal control elements (heaters, etc.). This shall include data as to when the element is ON/OFF or enabled, where its control sensor is located, what its set points are, control algorithm, etc.

An RTM presented in tabular form shall include the following groups of data:

Table 1: NODE IDs

Node #	Node Name	Node Description

Node Name = Acronym or short description of node
Node Des = More detailed node description

Table 2: NODE PARAMETERS 1

Node #	Node Type	Node Mass	Node SHC

Node Type = Diffusion, Arithmetic or Boundary
Node SHC = Node Specific Heat Capacity

Table 3: NODE PARAMETERS 2

Node #	Node Area	Node α_{BOL}	Node α_{EOL}	Node ε

Table 4: CONDUCTION (K) COUPLINGS

K-Conductor #	i	j	K_{ij}	(watts/C°)

Table 5: RADIATION (R) COUPLINGS

R-Conductor #	i	j	R_{ij}	(in^2)

Table 6: ELECTRICAL POWER DISSIPATIONS (SOURCES)

Node #	Q_i	(watts)

Table 7: ABSORBED EXTERNAL HEAT FLUX LOADINGS

Node #	S_i	ρ_i	μ_i	(watts)

S_i = Absorbed solar power

ρ_i = Absorbed earth-reflected solar power

μ_i = Absorbed earth-emitted IR power

Table 8: BOUNDARY NODE TEMPERATURES

Node #	T_i	(°C, K)

Table 9: NOTES/COMMENTS

Miscellaneous information relating to model

The model will be validated against a more detailed TMM such that the average internal temperatures agree to within ±5°C and interface heat fluxes (radiative and conductive) agree to within ±0.5 watts. At least 3 transient cases shall be used.

Surface models of the MHS shall be provided by MMS. The models shall be accompanied by sketches/diagrams which clearly identify and describe the surfaces (nodes) comprising each model.

The models shall be presented in TRASYS format or, if this is not possible or feasible, in the tabular form described below. Each of the surface models (whether in TRASYS format or in tabular form) shall be provided as a single file included on a 3.5 inch diskette. A printed listing of each file shall also be provided. In addition, MMS shall provide listings of the input files to their radiation coupling/heat flux program and the resultant outputs from that program.

The surface models shall be limited to no more than 40 nodes each (exclusive of any nodes representing their thermal environments).

The unit of length preferred by Astro Space is the "inch". Any other single unit, however, shall be acceptable. Radiation coupling factors derived from the surface models shall be expressed as areas - not as areas multiplied by the Stefan-Boltzmann constant.

A surface model presented in tabular form shall include the following groups of data.

Table 1: NODE IDs
- Node # — Node Name — Node Description
- Node Name = Acronym or short description of node
- Node Des = More detailed node description

Table 2: NODE PARAMETERS
- Node # — Node Area — Node α_{BOL} — Node α_{EOL} — Node ε

Table 3: NODE GEOMETRY
- Node # Geometric data for node relative to a local coordinate system

Table 4: NOTES/COMMENTS
- Miscellaneous information relating to model

The surfaces shall be defined relative to a local rectangular coordinate system whose axes are parallel to the spacecraft's coordinate system.

SI units shall be used.

3.3.1.2 Spacecraft Contractor

The spacecraft contractor will be responsible for enforcing the requirements and restrictions imposed on the thermal interface.

Interface hardware such as mounting brackets, reinforcement plates, cable insulation and multilayered insulation blankets used for interfacing purposes will be the responsibility of the spacecraft contractor.

The spacecraft contractor will define the interfaces the instrument has with the spacecraft and other instruments. This will include but not be limited to:

- geometries
- surface thermo-optical properties
- surface finishes (diffuse and specular)
- surface temperatures
- spacecraft and payload configuration

The spacecraft contractor will define the basic flux parameters to enable the instrument contractor to calculate the fluxes incident on the external surfaces of the instrument. The spacecraft contractor will define all external radiative couplings for all instrument surfaces.

The spacecraft contractor will provide protection against direct line-of-sight between pyrotechnics and the instrument thermal radiators as required.

The spacecraft contractor shall perform a detailed review of the Matra Marconi MHS thermal models. In support of this the spacecraft contractor shall interface with Matra Marconi and shall provide inputs to resolve any discrepancies in the Matra Marconi MHS thermal models. The spacecraft contractor will also compare the spacecraft model predictions to the resultant Matra Marconi instrument model predictions.

3.3.2 General Requirements

The instrument contractor shall be responsible for the thermal analysis and design required to satisfy the thermal requirements of the instrument.

The thermal design of the instrument shall provide for minimal thermal coupling between the instrument and the spacecraft structure (ESM). In particular, the net orbit-average energy transfer rate between the instrument package and the ESM shall not exceed the values shown in Figure 11A.

Thermal control of the instrument may utilize both passive and active elements.

3.3.3 Instrument Temperature Requirements

The allowable temperature ranges applicable to the instrument will be as defined in Table 9. The thermal control provided for the instrument will maintain the designated point-of-application temperature(s) within these ranges when the instrument is situated in the designated environment.

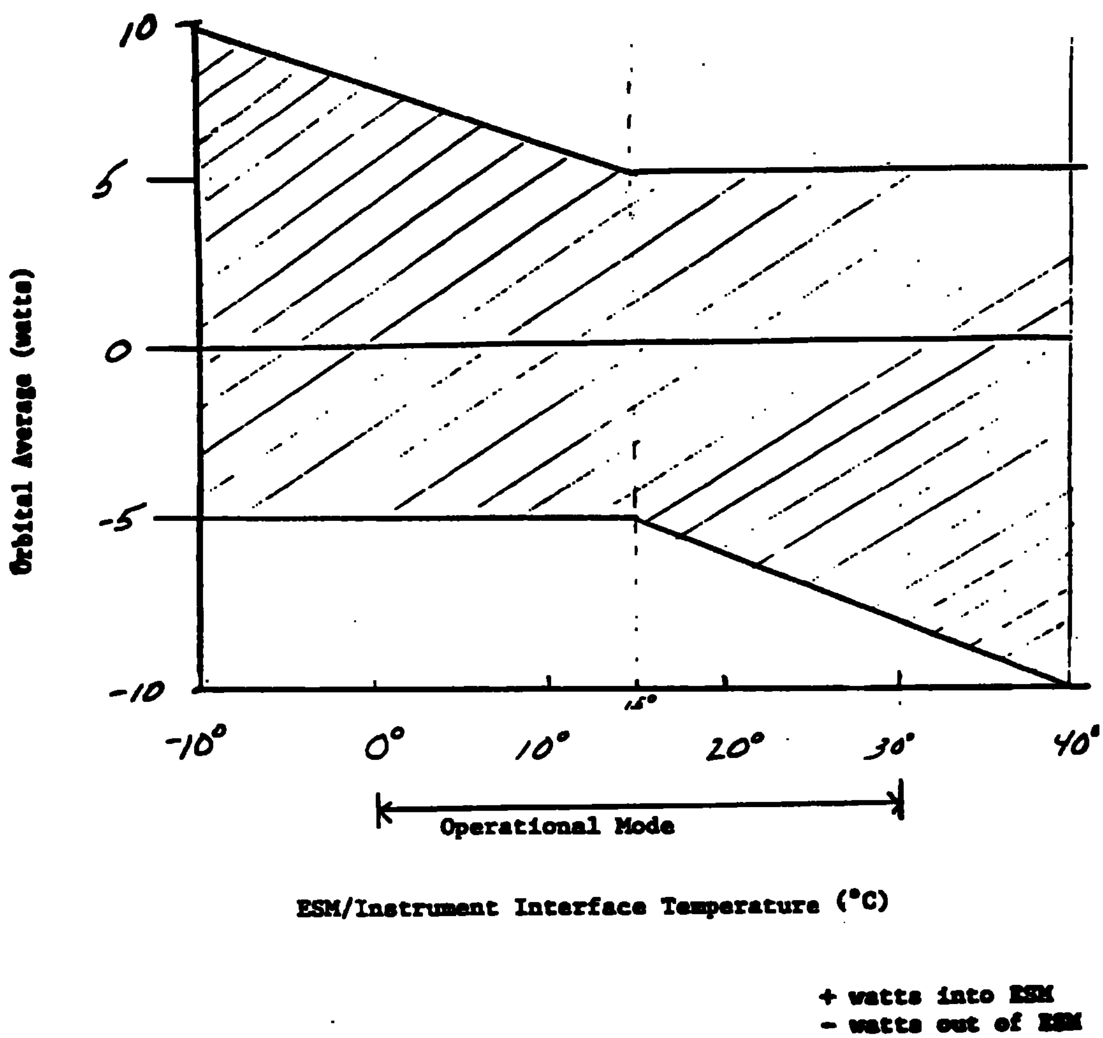


Figure 11A. Orbit-Average Energy Transfer

TABLE 9. INSTRUMENT ALLOWABLE TEMPERATURE RANGES

Temperature Range Definition		Range Limits (°C)		Application
		Min	Max	Point
(1)	Allowable on-orbit operating temperature range; instrument data within specification.	0	+36	Local Oscil.(H5 Temp)
(2)	Allowable on-orbit operating temperature range; instrument data not within specification.	-10	+45	Local Oscil.(H5 Temp)
(3)	Allowable on-orbit non-operating temperature range; (survival range).	-40 -30 -40	+60 +60 +60	Receiver Electronics Scan Mechanism
(4)	Allowable on-orbit MIN/MAX turn-ON temperatures.	-25*	N/A	Electronics
(5)	Allowable in-air long term storage temperature range.	-40	+60	Baseplate

*Corresponds to the minimum acceptance start-up temperature.

3.3.4 Spacecraft (ESM) Temperature Specifications

The spacecraft component (ESM) of the thermal interface is temperature-characterized as follows.

3.3.4.1 Operational Conditions

Orbital Sun-Angle	*Mean Interface (ESM) Temperature (°C)*	*Orbital Variation (C°)*
0°	13	±1
27.5°	19	±3
80°	23	±3

The interface temperature will be within ±5 degrees C of the mean value shown. The maximum rate of change of this interface temperature will not exceed 5°C per hour at any time on-orbit.

3.3.4.2 Survival (Safestate) Condition

Orbital Sun-Angle	*Mean Interface (ESM) Temperature (°C)*	*Orbital Variation (C°)*
0°	5	±3
27.5°	10	±3
80°	10	±3

The interface temperature will be within ±5 degrees C of the mean value shown. The maximum rate of change of this interface temperature will not exceed 5°C per hour at any-time on-orbit.

3.3.4.3 Orbits and Radiative Flux Parameters

3.3.4.3.1 Orbits

The spacecraft orbit utilized for analysis and design shall be a 450 nautical mile, sun-synchronous, circular orbit. The orbital sun-angle range will be 0° to 80°.

The orbital sun-angle (aka γ or γ-angle) may be defined as the angle between the spacecraft's positive orbit normal and a vector "to the sun". Clearly, $0° \leq \gamma \leq 180°$ and, for $\gamma = 0°$, the sun's rays are perpendicular to and "beating down" onto the orbital plane, while for $\gamma = 90°$, the sun lies in the plane. Refer to Figure 11B.

When on-orbit and at nominal attitude, the spacecraft's +Z-axis coincides with the positive orbit normal. Thus, the γ-angle also indicates the angle between the spacecraft's +Z-axis and the sun's rays. The angle is essentially constant over a single orbit, but varies in the near term due to drifting of the orbit's nodes and to changes in the solar declination. As the spacecraft traverses its orbit, therefore, the sun appears to "cone" about the spacecraft's +Z axis with half-angle γ and at a uniform azimuthal rate of 360° per orbital period. Specification of an azimuthal angle (along with the γ-angle) completely defines the direction of insolation on the spacecraft. A commonly referenced azimuth angle (known as ASOLANG) is available via spacecraft telemetry.

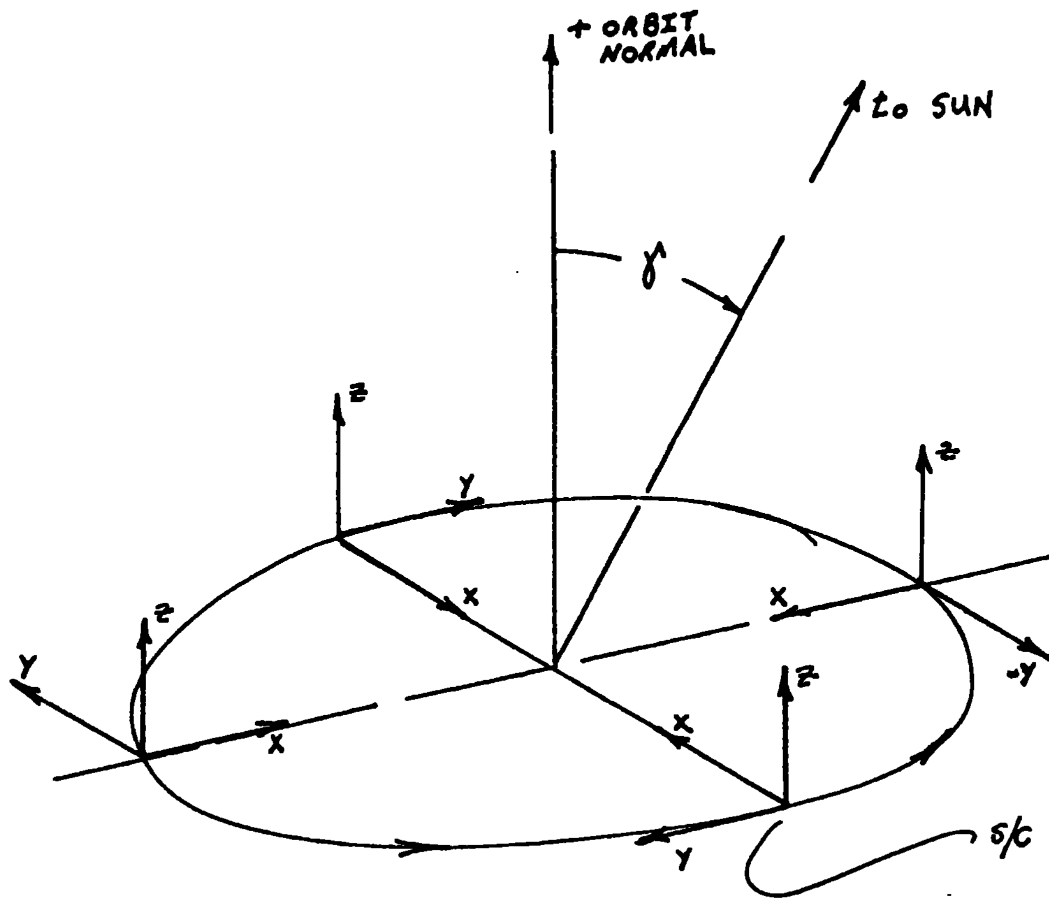


Figure 11B. The Sun (GAMMA) Angle

At nominal spacecraft altitudes, the orbit is free of earth eclipse for $0° \leq \gamma \leq 27.8°$. For $27.8° \leq \gamma \leq 90°$, eclipses occur, and their durations increase with increasing γ. Maximum eclipse time for the ATN-KLM mission ($\gamma = 80°$) is approximately 35 minutes.

3.3.4.3.2 Radiative Flux Parameters

The recommended values of parameters to be used for environmental flux determinations are given below:

	Minimum Value	*Maximum Value*
Solar Constants	1324.1*	1418.1*
Earth IR Radiance	213.8*	258.2*
Sub-Solar Point Albedo Radiance	358.8*	531.9*
Earth Albedo	0.271	0.375

Earth Radius	6378.145 km
Gravitational Parameter	3.986012×10^5 km^3/sec^2

*watt/m^2

3.3.5 Instrument Thermal Control Components

The following passive and active thermal control elements shall be incorporated within the instrument.

3.3.5.1 Passive Control Elements

3.3.5.1.1 Surface Finishes (External) and Fixed Area Radiators

Details for these items will be as specified in the following documents:

MHS Thermal Interface Control Drawing, MHS-ID-GA002-MMP

3.3.5.1.2 Multilayered Insulation Blankets

Details for these items will be as specified in the following document:

MHS Instrument Integration Constraints Document, MHS-TN-JA271-MMP

3.3.5.1.3 Mounting

Instrument mounting details will be as specified in the following document:

Instrument Configuration Document, MHS-ID-JA029-MMP

3.3.5.2 Active Control Elements

3.3.5.2.1 Operational Heaters

The operational heaters will be used within the Receiver and the Scan Mechanism. These will not be used during normal operational mode. They will be used during warm-up. These heaters will be controlled by the instrument software.

3.3.5.2.2 Louver/Radiator Assemblies

NONE

3.3.5.2.3 Survival Heaters

Survival heaters will be used within the Electronics, the Receiver, the Scan Mechanism, and the instrument baseplate and will be thermostatically controlled. These heaters will maintain minimum survival temperatures, except for the Electronics heater which will maintain minimum turn-on temperature.

It is not permitted to supply power simultaneously on the survival bus interface and on the safety heater test interface. The MIU will provide protection diodes on the survival heater bus (because the survival heaters will also be used as safety heaters during spacecraft thermal vacuum testing).

3.3.6 Test Instrumentation

3.3.6.1 Test Safety Heaters

The MHS survival heaters will be used as safety heaters. A test connector will be used to power these from a separate source. These heaters shall be powered from sources external to the spacecraft bus.

3.3.6.2 Test Thermocouples

The instrument contractor will not provide test thermocouples. The spacecraft contractor may install thermocouples for use during spacecraft-level thermal/vacuum testing. The location and attachment of the test thermocouples will be mutually agreed upon by the spacecraft and instrument contractors.

3.4 Environmental Interface

3.4.1 Magnetic Interface

3.4.1.1 Spacecraft-Generated Magnetic Fields

The spacecraft will generate magnetic fields during orbital operations. The maximum unloading coil fields will be generated during angular momentum unloading which occur in certain regions of the orbit as defined below:

Spacecraft Axes to be Unloaded	*Geomagnetic Latitudes of Unloading Regions*
Pitch	Within ±24° of the Geomagnetic Poles
Roll Only	Within ±24° of the Geomagnetic Poles
Roll and Yaw Combined	Within ±17.5° of 37.5° Geomagnetic North Latitude or Within ±17.5° of 37.5° Geomagnetic South Latitude

The maximum magnetic flux the instrument will experience on the spacecraft will be 2.0 gauss. This level will be considerably reduced in orbit through the use of magnetic moment compensation.

The roll/yaw torquing coil is perpendicular to the spacecraft pitch (Z) and is attached to the -Z end of the ESM. The pitch torquing coil is perpendicular to the spacecraft roll (Y) axis and is attached to panel #3 of the ESM. The torquing coil parameters are as follows:

Parameter	*Pitch Coil*	*Roll/Yaw Coil*
Length (in)	53**	34
Height (in)	17.5	23.3*
Dipole (ATM^2)	21.4	38.6

*Height of Rectangle of Equivalent Area.
**Assume Coil is centered on Panel #3 of the ESM.

3.4.1.2 Instrument Magnetic Properties

Instrument magnetic properties shall conform to the following requirements. Note: Dipole moments, when specified, are as though determined from magnetic field measurements conducted at a distance of three times the maximum linear dimension of the item under test.

a. Initial Perm Test. The maximum D.C. magnetic moment of the instrument following manufacture shall not exceed 0.2 ampere-meters^2.

b. Perm Levels After Exposure to Magnetic Field. The maximum magnetic moment of the instrument after exposure to magnetic field test levels of 15 x 10^{-4} tesla shall not exceed 0.3 ampere-meters^2.

c. Perm Levels After Exposure to Deperm Test. The maximum magnetic momentum of the instrument after exposure to magnetic field deperm levels of 50 x 10^{-4} tesla shall not exceed 0.1 ampere-meters^2.

d. Induced Magnetic Field Characteristic Measurement. The induced magnetic field of the instrument shall be measured while the instrument is turned off and exposed to a magnetic field test level of 0.6 x 10^{-4} tesla. The measurement shall be made by a test magnetometer that can null the test magnetic field. The results of this test shall be recorded in the instrument test data record.

e. Stray Magnetic Field Measurement. The instrument magnetic moment shall not change in excess of 0.05 ampere-meters^2 due to internal current flows.

3.4.1.3 Instrument Degaussing

The MHS will not be degaussed.

3.4.1.4 RCE Latch Valve Stray Magnetic Field

The latch valve contains two permanent magnets with dipole moments as follows: Open Position; MX = -0.1 ATM^2, MY = -1.04 ATM^2, MZ = 0.025 ATM^2: Closed Position; MX = 0.06 ATM^2, MY = -1.08 ATM^2, MZ = 0.06 ATM^2. The latch valve is located on the RSS approximately nine (9) inches from the spacecraft separation clamp and 10° off the +X axis towards +Y.

3.4.2 Electromagnetic Interference (EMI)

On the TIROS spacecraft, the instrument will be exposed to the levels defined in Table 10.
The instrument shall be compliant with the Radiated Emission requirements defined in section 3.5.2.2 of GSFC-S-480-53.

3.5 Operational Requirements and Precautions

3.5.1 Storage Requirements

These are requirements for instrument storage at the spacecraft contractor's facility and are the responsibility of the spacecraft contractor.

3.5.1.1 Storage in Instrument Transit Case

(1) General: The MHS will be installed and stored in its transit case.

(2) Temperature Limits: 75 ±10°F

(3) Humidity Limits: 35% to 55%

TABLE 10. RF FIELDS AT MHS INSTRUMENT

Spacecraft Antenna	Instrument Antenna (v/m)	Frequency (MHz)
SBA-1	20.6	1698
SBA-2	18.4	1702.5
SBA-3	27.9	1707
SLA	19.5	1544.5
VRA	11.5	137.5/137.62
SOA(1)	8.3	1702.5
SOA(2)	7.1	2247.5
BDA	6.4	137.35/137.77

(1) Earth-facing antennule. Assumed to be co-located with the earth-facing antennule of another S-Band omni - the one for the new S-Band beacon, at an unspecified frequency. Normally only the beacon omni will radiate when the instrument is on. For both omnis radiating, assume each one produces the given field strength, and omit radiation from SBA2.
(2) Earth-facing antennule of launch/emergency omni.
(3) Field strength calculations are at the center of the instrument antenna.

(4) Storage Pressure: The MHS shall be stored in its transit case. The spacecraft contractor can provide pressurized air or nitrogen periodically as required.

(5) Other: The MHS shall be bench tested at least once every 12 months with the exception that it must have been tested within 4 months before being removed from storage for installation on the spacecraft.

3.5.1.2 Storage-Mounted on Spacecraft

(1) Storage of the instrument when mounted on the spacecraft shall be at Class 100,000 or better.

(2) Humidity Limits of Storage Area: Maximum relative humidity 55%, minimum relative humidity 35%. Spacecraft may be stored in a tent purged with LN_2 boiloff.

(3) When spacecraft is stored with instruments installed, periodic spacecraft electrical tests will be performed each nine (9) months which will include a brief instrument electrical check.

(4) Temperature Limits: -0°C to +40°C

3.5.2 Test Requirements

(1) Test Constraints:

The MHS test constraints are defined per MHS Integration Constraints Document, MHS-TN-JA271-MMP

(2) Handling: Controls and procedures shall be per the following documents (instrument contractor supplied):

MHS Integration Constraints Document, MHS-TN-JA271-MMP

(3) Temperature Limits:

Refer to Table 9.

(4) Cleanliness: All spacecraft tests will be performed in a Class 100,000 clean room environment except for acoustics, pyro shock, T-V preparations, and during transportation. For these tests the instrument vendor can provide protective covers or instructions how to bag them with a protective film.

3.5.3 Operational Requirements

3.5.3.1 Command Sequences

Command sequences for MHS operations will be as follows:

3.5.3.1.1 Initial Turn-On Sequence (In-Orbit)

Initial Turn-On Sequence (In-Orbit) shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.1.2 Normal Turn-On Sequence (Test)

Normal Turn-On Sequence (Test) shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.1.3 Turn-Off Sequence

Turn-Off Sequence shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.1.4 Safestate Sequence

1. Send the "safeing" command to the MHS via the command/telemetry 1553 interface (refer to section 4.2.1.1 of the MHS TM-TC and Science Data Format Directory, MHS-TN-JA063-MMP for details).

   This command will require a maximum of three seconds to execute and park the MHS reflector pointing at the on-board target. Upon completion of this command, the MHS will be waiting for the instrument power to be switched off.

2. Switch off the MHS main power and pulse load power buses; and switch on the MHS survival power bus.

Safe State Sequence shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.2 Turn-On Constraints

Turn-On Constraints shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.3 Initial In Orbit Turn-On Constraints

Initial In Orbit Turn-On Constraints shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.4 Sync Interruption

When the 8 second sync pulse stops, the MHS will continue to operate nominally provided the Reference Clock is present, with the instrument timing based on the last 8 second sync pulse received.

Subsequent 8 second sync pulse datums within the instrument will occur at 8 second intervals derived from the Reference Clock. If the Reference Clock is removed, the MHS will enter Fault Mode irrespective of whether the 8 second sync pulse is present or not. In this mode all science data gathering and scan control activities are stopped and the instrument needs to be commanded to restore nominal operation.

If the 8 second sync pulse is reapplied such that it is not in phase with the previous 8 second sync pulse, then the MHS will immediately start to resynchronize to the new 8 second sync pulse. However, there will be a significant disturbance to all functions, e.g., scan control, science data gathering, generation of housekeeping and science data packets. All housekeeping and science data will be invalid for one to two scan periods, but after this time the MHS will recover and resume nominal operation, with no external commanding necessary. The internal instrument software is not expected to crash.

Sync Interruption response shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

3.5.3.5 Launch Configuration

Launch Configuration shall be as defined per MHS Flight Operations Manual, MHS-OM-JA215-MMP.

There are no restrictions imposed by MHS on the solar aspect angle during the launch/ascent phase.

3.5.4 Transportation of Instrument While Mounted on Spacecraft

The environments experienced during spacecraft transportation and handling will be controlled to be significantly less severe than worst case flight conditions. During transportation and handling the spacecraft may temporarily be in areas that do not meet class 100,000. During these periods the spacecraft will be in a protective tent. When this tent cannot be used, the instrument will be bagged to prevent its being contaminated.

4.0 INSTRUMENT INTEGRATION, TEST AND OPERATING REQUIREMENTS AND CONSTRAINTS

4.1 Test Equipment and Service

4.1.1 Equipment to be Supplied by Instrument Contractor to the Spacecraft Contractor

The following is an inventory of the equipment that shall be supplied by the instrument contractor:

(a) Electrical Ground Support Equipment (EGSE) - This unit shall be capable of operating the instrument in all its operating modes.

(b) Contamination Covers - This cover(s) shall be used to minimize the accumulation of contamination on optical surfaces both during the bench check testing and while the instrument is on the spacecraft.

(c) Handling Fixture - The handling fixture shall be used to lift the instrument from its shipping container and shall also be used to handle the instrument during bench operations. It shall be removed from the instrument prior to installation of the instrument on the spacecraft unless the handling fixture is part of the lifting fixture.

(d) Thermal Blankets - The thermal blankets shall be as shown on sheets 2 and 3 of the MHS General Asembly, MHS-AD-JA001-MMP. These blankets shall be shipped fitted to each instrument.

(e) Connector Savers - A set of connector savers shall be provided with each instrument. These conncctor savers will remain on the instrument until it is integrated on the spacecraft and will not be removed until after the IPF is performed.

(f) Optical Alignment Equipment - A permanent alignment mirror(s) shall be provided.

(g) Thermal Vacuum Target and Controller - There will be one variable temperature (LN_2 to room temperature) target .

(h) Cables - The cables required to connect the EGSE to the instrument shall be supplied.

(i) Lifting Fixture - This fixture shall attach to the instrument and shall allow the instrument to be mounted to the spacecraft when the spacecraft is in a vertical position. The fixture shall have provisions for lifting with a crane.

4.1.1.1 Electrical Ground Support Equipment

The Electrical Ground Support Equipment (EGSE) is described in the EGSE User Manual 3175-34024-MAU.

4.1.1.2 Special Calibration Test Equipment

There will be one variable temperature (LN_2 to room temperature) target and controller. The target will be as defined in the SCTE User Guide, MHS104 (M/RSI/41/5/2).

4.1.1.3 Protective Cover

The protective cover consists of a single red-colored item. The mounting/dismounting and operation constraints are defined in the MHS Instrument Integration Constraints Document, MHS-TN-JA271-MMP.

4.1.1.4 Handling and Lifting Fixtures

The operation instructions for use of the handling and lifting fixtures shall be given in 3175-35032-MAU.

4.1.1.5 Thermal Blanket

The thermal blankets, which must be attached to the instrument after it is mounted on the spacecraft, shall be detailed in MHS Integration Constraints Document, MHS-TN-JA271-MMP.

4.1.2 Services Provided by Instrument Contractor at the Spacecraft Contractor Facility

4.1.2.1 Bench Test

The Bench Test and associated data analysis of the first Flight Model shall be performed by instrument contractor personnel. During the performance of this test the instrument contractor shall instruct the assisting spacecraft contractor personnel in the use of the EGSE and the performance of the bench test.

4.1.2.2 Data Analysis

There are no provisions to send instrument contractor personnel to spacecraft contractor to review data other than informally or in a trouble-shooting mode. Instrument contractor personnel will be present at the spacecraft contractor during further spacecraft test as directed by EUMETSAT.

4.1.2.3 Troubleshooting

Instrument Contractor and EUMETSAT personnel shall be available at the spacecraft contractor to assist in troubleshooting upon direction from EUMETSAT.

4.1.2.4 Warranty

There are no warranty provisions between EUMETSAT and the spacecraft contractor. If repair of an MHS is necessary NASA will arrange with the EUMETSAT to have it completed.

If the MHS must be shipped back to the vendor, the spacecraft contractor will assure the unit is packed in the original shipping container.

The spacecraft contractor will arrange all transportation, at the direction of NASA, and intermediate storage to conform to the storage environmental limits in this specification. If the instrument is to be shipped by air, spacecraft contractor will escort the shipment to the air terminal.

The spacecraft contractor transportation office will make these arrangements based upon the local climate at the time of shipment. If for some reason these environmental limits cannot be assured, the shipment will be held, and NASA will be notified.

4.1.2.5 Equipment Maintenance to be Supplied by Instrument Contractor

The instrument contractor will be responsible for maintenance of all test equipment delivered to spacecraft contractor as directed by EUMETSAT, except for any designated, commercial test instruments which will be maintained by spacecraft contractor.

Maintenance or repairs can be done at the spacecraft contractor, or, in the event any equipment needs to be shipped to the instrument contractor, the spacecraft contractor will accept responsibility for all transportation arrangements as defined in Paragraph 4.1.2.4 of this document.

4.1.3 Documentation to be Supplied by the Instrument Contractor to the Spacecraft Contractor

4.1.3.1 Bench (Functional) Test Procedure

The Bench (Functional) Test Procedure will be supplied to the Spacecraft Contractor concurrent with delivery of the EGSE. The preliminary versions of this procedure will be submitted to the Spacecraft Contractor as generated.

4.1.3.2 GSE Operations Manuals and Procedures

The ancillary manuals and procedures necessary for use of the various test equipment will be shipped to Martin Marietta concurrent with delivery of the EGSE; preliminary version of these documents will be submitted as generated. The documents covered by this paragraph are:

(a) 3175-34024-MAU, EGSE User Manual

(b) 3175-35032-MAU, Lifting and Handling Devices User Manual

(c) 3175-35042-MAU, Cryostat User Manual

(d) 3175-35043-MAU, Transportation Container

(e) MHS104 (M/RSI/41/5/2), SCTE User Guide

4.1.3.3 Data Book, Specification Verification and Calibration

The following data will be supplied with each instrument. The alignment portion shall contain:

(a) Instrument Alignment to its mounting surface;
(b) Optical Fields of View;
(c) Size, Weight and Center of Gravity of each module.

The calibration data shall comprise:

(a) Conversion equations for housekeeping and science telemetry
(b) Conversion Equations for Analog telemetry

4.1.3.4 Handling Procedures

The MHS Integration Constraints Document, MHS-TN-JA271-MMP, will be delivered with the first Flight Model. Preliminary versions of this document will be submitted to the spacecraft contractor as generated.

4.1.4 Equipment and Services to be Supplied by the Spacecraft Contractor for Direct Instrument Support

4.1.4.1 Spacecraft Contractor Supplied Equipment and Services

a. Power Input at Test Location

1. 110 Vac, 60 Hz, Single Phase, 30 Amp. Service for EGSE

b. Floor Space to Accommodate the Following (GFE) Equipment:

	Size (Inches)	Weight (Pounds)
1. Electrical Ground Support Equipment	TBS	TBS
2. Special Calibration Test Equipment	TBS	TBS

c. Test Area

The contractor will provide a test area for the MHS which meets the following environmental requirements:

1. Cleanliness - Class 100,000
2. Temperature Limits - 65°-85°F (18 - 29°C)
3. Relative Humidity - 35 to 55 Percent

d. Standard Test Equipment

Standard lab test equipment will be available at the spacecraft contractor facility to set up and troubleshoot the EGSE and to use during the bench test and post storage tests. LN2 will be available during bench testing and thermal-vacuum test preparation.

4.1.4.2 Spacecraft Contractor Supplied Labor for Testing at the Instrument Level

a. First Flight Model - The Spacecraft Contractor will assist the Instrument contractor personnel in performing the Bench Test.

b. Flight Models and Post Storage Testing - Martin Marietta will perform the Bench Tests. The bench test will be performed at nine (9) month intervals on instruments not mounted on a spacecraft.

4.1.5 Test Access to the MHS

4.1.5.1 During Bench Checkout

All electrical interfaces to the instrument will be through the MHS/Spacecraft connectors. There may be test connectors on the MHS but these are to be used only during the Bench Checkout; access to them will not normally be required and is not allowed unless authorized by the instrument contractor.

4.1.5.2 During Satellite Level Tests

Access will be required to the instrument during the spacecraft level testing to remove the protective cover over the antenna. The dust cover will be kept on the scan cavity during all testing except the (1) EMI test, (2) vibration and (3) during the thermal vacuum test.

4.1.5.3 Access for Inspecting Scan Antenna

Access to the instrument will be required just before enclosing the spacecraft with the shroud for the purpose of visually inspecting the antenna.

4.1.5.4 During Launch Pad Testing (Fairing On)

There shall be no need for visual inspection of the instrument on the Launch pad. There will be no targets mounted in the fairing for the instrument use.

4.2 Acceptance Test Performed at the Instrument Contractor

The tests that are to be performed by the Instrument Contractors are defined in the EUMETSAT Specification for the MHS (EPS/MHS/SPE/93001).

4.3 Testing at the Spacecraft Contractor Facility

The objective of testing the instrument at the Spacecraft Contractor Facility is to assure compatibility of the Spacecraft Contractor's MHS Interface Unit (MIU) with the instrument, the instrument with the spacecraft, and to demonstrate that the instrument meets its specified characteristics. Prior to receipt of the MHS flight models by the Spacecraft Contractor, the Spacecraft Contractor shall demonstrate compatibility of the MIU by performing a compatibility test with the MHS Engineering Model (EM). Before conducting the test, the Spacecraft Contractor shall prepare a test plan. The Instrument Contractor shall provide support data as required by the Spacecraft Contractor, the electrical ground support equipment (EGSE), the MHS EM, and engineering support for the test. The Spacecraft Contractor will also supply MIU data required by the Instrument Contractor for MHS interface and proper conduct of the test. The spacecraft/instrument test program is divided into tests performed on the instrument only, i.e., the Instrument Evaluation Tests; and on the instrument as part of the spacecraft system, i.e., the System Evaluation Tests and the Environmental Tests. The test flow of this is given in Figure 12. Test failures related to the instrument will be documented by a Test Discrepancy Report per operating instruction PAP E8.14 as specified in the Quality Assurance Plan, 3267412. These tests will be performed by the spacecraft contractor, with the exception of the post-delivery (incoming) electrical bench (functional) test, which will be performed by the instrument contractor.

4.3.1 Instrument Evaluation Tests

The objective of the Instrument Evaluation Tests is to demonstrate that the instrument has the same characteristics at the spacecraft contractor as it did when tested at the instrument contractor's plant before shipment. At the completion of the Evaluation Tests the instrument is either put on the spacecraft or is put into storage to wait for later mounting. This evaluation test is divided into Receiving, Incoming Inspection Mechanical, and Incoming Inspection Electrical. Storage and storage retesting are also considered part of the Instrument Evaluation Tests.

4.3.1.1 Receiving

The objective of the receiving tests and inspection is to detect any gross damage during shipping and to verify delivery of documentation supplied with the instrument. The transit package will not be opened during receiving inspection. Alignment and calibration and other instrument related data will be reviewed by the spacecraft contractor Systems Engineering.

4.3.1.2 Incoming Inspection - Mechanical

The objective of the mechanical incoming inspection is to check for physical damage to the instrument and to document its condition as received at the spacecraft contractor. The state of the shock indicators will be determined and recorded. The instrument will be weighed. This weight will be used in establishing the full spacecraft weight.

4.3.1.3 Degaussing

The MHS will not be degaussed.

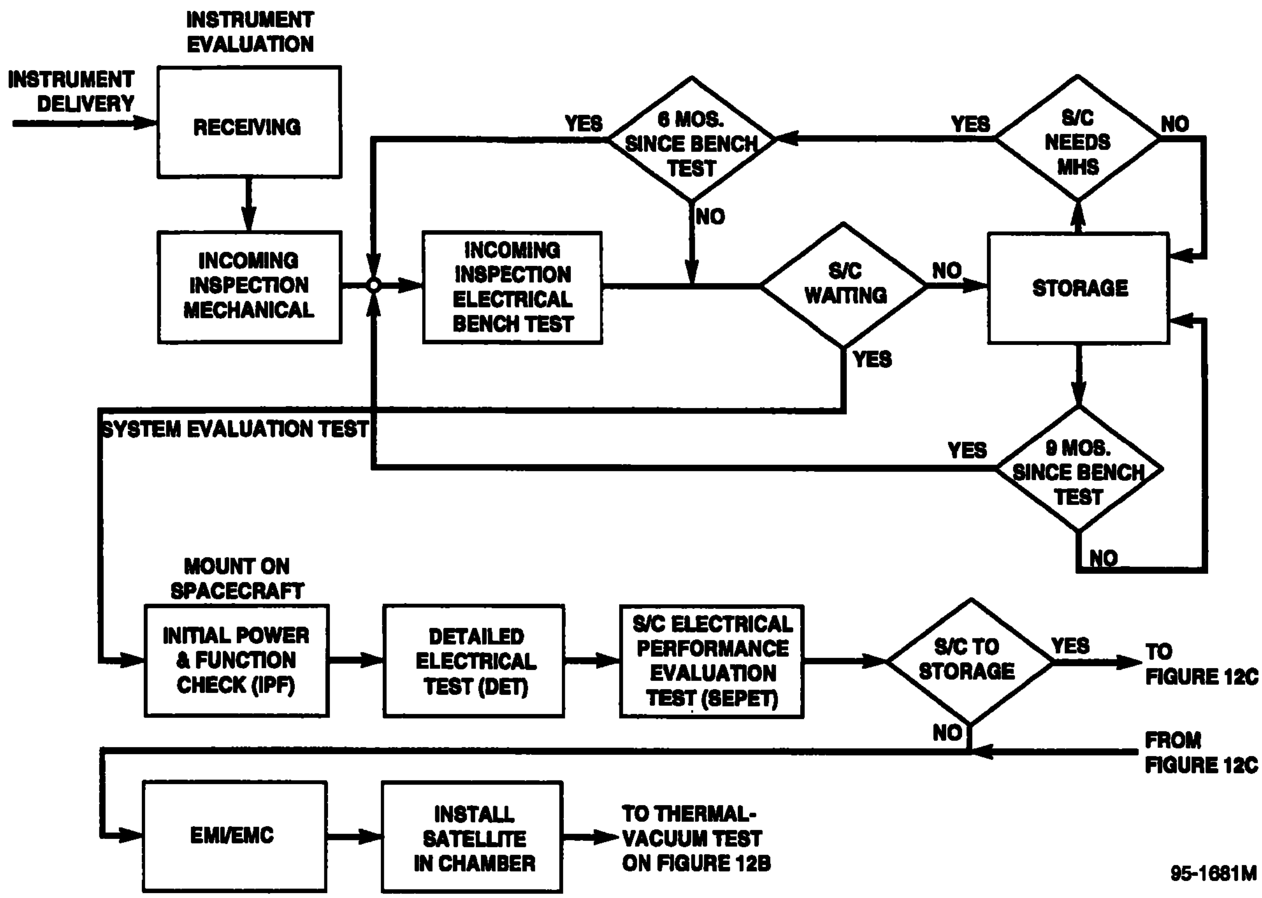


Figure 12A. MHS Testing at Spacecraft Contractor

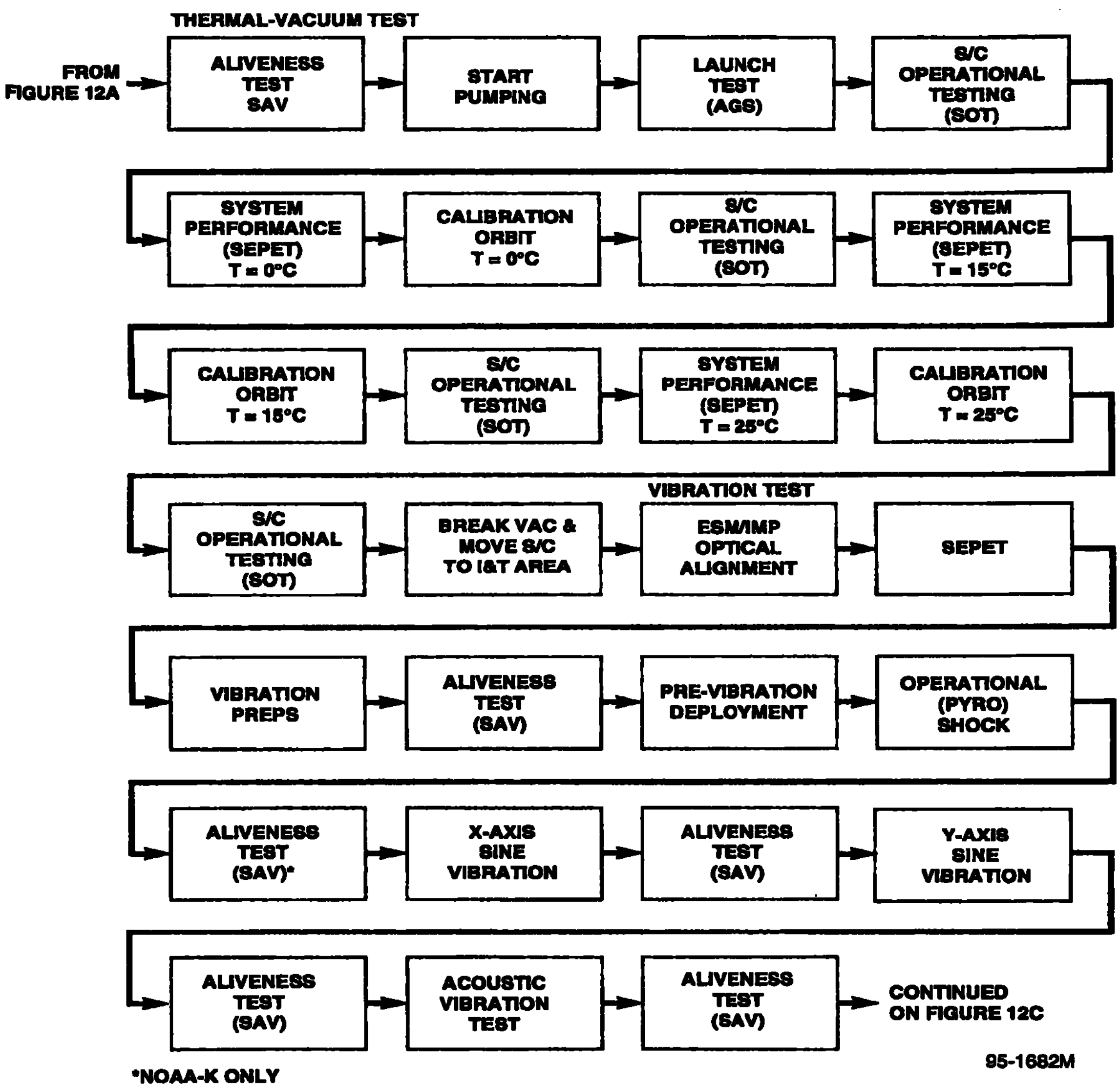


Figure 12B. MHS Testing at Spacecraft Contractor (Continued)

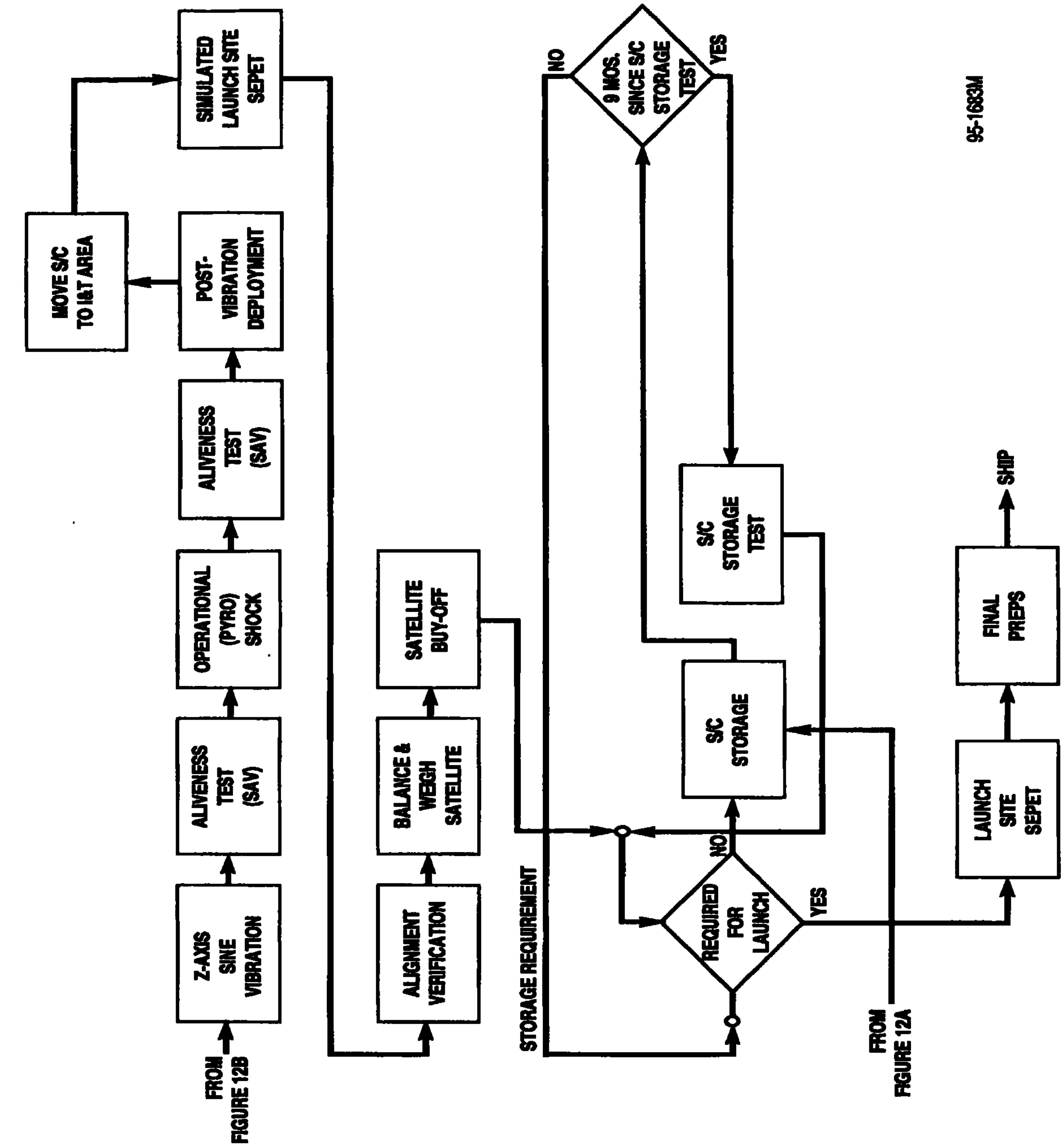


Figure 12C. MHS Testing at Spacecraft Contractor (Continued)

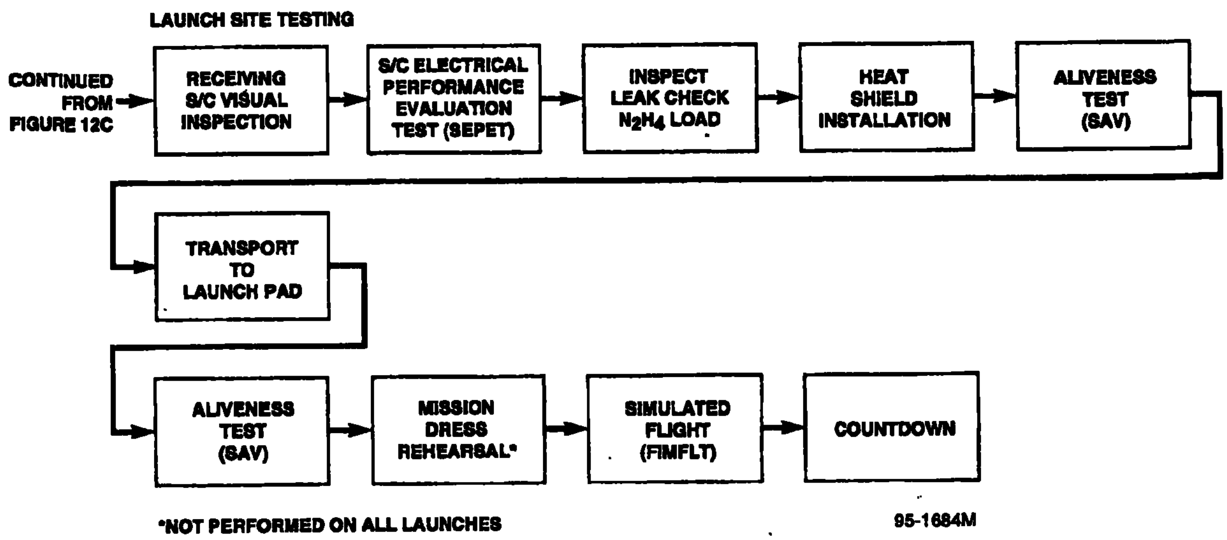


Figure 12D. MHS Testing at WTR

4.3.1.4 Incoming Inspection - Electrical (Bench Test)

The Bench Test will be performed to ensure that the electrical and functional characteristics have not changed as a result of shipping. The Electrical Ground Support Equipment (EGSE) will be separately tested before being connected to an instrument.

4.3.1.5 Storage and Storage Testing

The MHS will be stored following the Incoming Electrical Inspection if not integrated on a waiting spacecraft.

The purpose of Storage testing is to assure that the instrument has not failed during storage. Instruments in storage shall be tested nine (9) months after the last bench test and every nine months thereafter. These periodic tests are comprised of a Bench Test.

Instruments which have been in storage more than six (6) months will undergo a Bench Test before installation on the spacecraft. The requirements are given in Paragraph 3.5.1 of this specification.

4.3.1.6 Instrument Test Matrix

The instrument level and spacecraft level test matrix is given in Tables 11A and 11B.

4.3.2 Mounting to Spacecraft

When the MHS is to be mounted on the spacecraft, it will be placed on a clean bench and the handling fixture will be removed. The lifting fixture, after having been cleaned, will be attached to the instrument. The scan cavity dust covers will be checked for proper installation. The MHS will then be installed on the spacecraft.

4.3.3 System Evaluation Test

The objectives of the System Evaluation Test are to integrate the instrument to the spacecraft system and to assure that the MHS meets all interface requirements.

The System Evaluation Test is divided into the Initial Power and Functional Check (IPF); the Detailed Electrical Test (DET); and the Spacecraft Electrical Performance Evaluation Test (SEPET).

For all tests the instrument is mounted on the spacecraft and the test data can be processed by the ATNAGE.

TABLE 11A. MHS TEST MATRIX

Test	Bench Test	System Evaluation			Aliveness	Thermal Vacuum		
		IPF	DET	SEPET		AGS	SEPET	Cal Check
Reference Paragraph	4.3.1.4	4.3.3.1	4.3.3.2	4.3.3.3	4.3.4.1		4.3.4.2	
Power Status S/C PWR OFF	X							
S/C ON, MHS OFF						X		
MHS ON		X	X	X	X		X	
1. Ground Resistance Measurements		X						
2. Harness Verification		X						
3. Power Measurements	X	X						
4. Input Signal Level Measurement	X							
5. Command/ Mode Verification	X		X	X	X		X	
6. Output Signal Level Measurement	X		X					
7. Data Format Verification	X		X	X	X		X	
8. Limit Check Data	X		X	X	X		X	
9. Limit Check Analog Telemetry	X	X	X	X	X	X	X	

ITAR CONTROLLED DATA

TABLE 11B. MHS TEST MATRIX

Test	Environmental Tests				
	Optical Alignment	Sine Vibration	Acoustic Vibration	Post-Vib Deployment	Final SEPET
Reference Paragraph	4.3.4.3	4.3.4.4	4.3.4.5	4.3.4.6	4.3.4.7
Power Status S/C PWR OFF	X				
S/C ON, MHS OFF		X	X	X	
MHS ON					
1. Ground Resistance Measurements					
2. Harness Verification					
3. Power Measure					
4. Input Signal Level Measurement					
5. Command/Mode Verification					X
6. Output Signal Level Measurement					
7. Data Format Verification				X	
8. Limit Check Data				X	
9. Limit Check Analog Telemetry	X	X	X	X	

4.3.3.1 Initial Power and Functional Checks (IPF)

The objectives of the Initial Power and Functional Checks are (1) to provide an orderly method of verifying that application of power to the MHS will not damage it or previously integrated subsystems; and (2) to verify that, after mating the correct electrical interface has been established.

Correct operation of the instrument will be established by the use of breakout boxes and probes as required. Input signal voltages and power level measurements will be made on the spacecraft harness prior to mating with the MHS. Breakout boxes and/or probes may be used to expedite the measurement of signal levels and loading in all operational modes. However, the use of any breakout box or probe at the MHS/spacecraft interface will be subject to the following provisions: (1) all voltage taps will be protected against damage by external shorts pin-to-pin and pin-to-ground and no breakout box will contain more than one MHS connector; and (2) all power level or current measurements will be made using clip-on induction probes and extender harnessing. The parameters to be tested during the IPF are the following:

(a) Case Grounding - Verify case ground is firmly attached to the spacecraft ground.

(b) Harness verification - These measurements verify that electrical inputs to the instrument are on the correct pins. The presence of power on input pins, both full time and switched, is verified. Command functions on assigned lines are verified. Clock signals on assigned lines are verified. This harness verification is done prior to the initial instrument installation. If the MHS is exchanged the harness verification will not be repeated. Resistance of all MHS/Spacecraft Harness ground return lines will be measured.

(c) Instrument ground isolation - All power supply and signal grounds will be checked for isolation from the spacecraft ground before the spacecraft harness is connected.

4.3.3.2 Detail Electrical Test (DET)

The purpose of the Detailed Electrical Test is to demonstrate that the correct interface exists between the instrument and the spacecraft. The correct interface may be demonstrated by using breakout boxes (subject to the constraints of Paragraph 4.3.3.1) to measure signal levels at the interface. The DET will include a functional checkout in which the instrument is commanded into each of its states to verify correct electrical and mechanical response. The parameters to be tested during the DET will be the following:

(a) ***Input Signal Level Measurement*** - The measurements of the clock signal and command signal levels are made to ensure that the instrument is supplied the correct amplitude signal and that the input of the instrument does not load down the driving circuit.

(b) ***Command/Mode Verification*** - Verification of the correct response to each command will be measured by the mechanical response and electrical output.

(c) ***Output Signal Level Measurements*** - The measurement of the output signal levels will be made to ensure that the instrument is supplying the correct levels to the spacecraft and the spacecraft does not incorrectly load the circuits. The measurement will also verify that signals exist at each of the outputs of the housekeeping and Analog Telemetry which corresponds to the operating function.

(d) ***Scan Operation Verification*** - The correct scan operation will be verified using the information output contained in the housekeeping data stream.

(e) ***Data Format Verification*** - Verification of the Data Format will be demonstrated by the ability of the ATNAGE to decode and display all of the MHS words contained in the data stream. In addition this test will validate the database being used with the instrument.

(f) ***Analog Telemetry Verification*** - Verification of the operation of each of the Analog Telemetry points will be demonstrated by ATNAGE limit checking. This test will also verify the validity of the data base.

4.3.3.3 Spacecraft Electrical Performance Evaluation Test (SEPET)

The SEPET has two basic objectives: (1) to demonstrate by measurement that the system meets all specification criteria; (2) to compare the data with previous measurements or establish the basis for future comparison. The SEPET is the most comprehensive ambient electrical test of the entire spacecraft.

The SEPET will be performed in a room temperature environment at atmospheric pressure.

(a) Command/Mode Verification - Verification of correct response to each command will be measured by the mechanical response and electrical output. Status verification will also be performed using telemetry.

(b) Limit Check - A number of housekeeping telemetry words will be continuously limit checked in real time.

(c) Limit Check Analog Telemetry - All analog telemetry will be limit checked.

(d) Scan Verification - This test will verify the correct operation of the scan motor. The MHS scan verification shall be conducted in accordance with the test constraints identified in Section 3.5.2.1.

4.3.4 Satellite Environment Test

4.3.4.1 Aliveness Test

The aliveness test verifies that the instrument is correctly set up to enter a specific environment or has successfully passed the environmental exposure. The evaluation is accomplished by commanding the instrument through its orbital modes and status checking the data.

4.3.4.2 Thermal-Vacuum Tests

The purpose of the Thermal-Vacuum Test is to demonstrate the successful performance of the integrated satellite at temperature extremes in a vacuum environment.

The test will be performed with the spacecraft in the 24 foot or 35 foot vacuum chamber at the spacecraft contractor facility. The pressure will be less than 10^{-5} TORR and the walls of the chamber maintained at -65°C.

During the Thermal-Vacuum operation, the tests will be divided into AGS (launch simulation), Aliveness, System Performance, and Calibration check orbits. The functions to be tested in each of these tests are shown in Table 9. Details of the tests are given below:

(a) AGS - A simulated launch test will be performed following pumpdown. The MHS will be in its Launch mode.

(b) T-V SEPET - The test has similar objectives to the SEPET performed in the ambient condition.

(c) Calibration Check Orbits - The MHS will be in a normal test mode. No special testing will be performed during the calibration tests.

(d) Transition Test - There will be no instrument testing during the time spacecraft temperatures are adjusted from one plateau to the next.

    However, the normal status monitoring and data processing will be done during the period of temperature transition.

4.3.4.3 Optical Alignment

The purpose of this test is to determine the field-of-view of the instrument with respect to the satellite primary reference, the Earth Sensor Assembly (ESA). The measurement made during this test will be to determine the differences in pointing direction of the surfaces of an optical mirror(s) mounted on the instrument and the axes defined by mirrors on the AMSU-A1. The instrument will be mounted so as to meet the placement requirement of ±0.05 degrees in the Y- and Z-axes relative to AMSU-A1. This measured difference will be added to vendor supplied data which references the fields-of-view and axis to the instrument mounted cube coordinates.

4.3.4.4 Sine Vibration in X, Y, and Z Axes

The purpose of the sine vibration is to demonstrate the adequacy of the integrated spacecraft structure design. Results of the test will be used to verify the major critical resonances and the adequacy of the individual components to withstand vibration in each of three orthogonal axes. The spacecraft will be vibrated in an all up flight configuration.

The MHS will be in its launch configuration. Some limited analog telemetry will be on during vibration. The scan cavity dust covers will be removed for this test.

Between each axis of vibration the instrument will be inspected. An aliveness check will also be performed between vibration axes as shown in the flow of testing at the spacecraft contractor facility (Figure 12).

4.3.4.5 Acoustic Vibration

The purpose of the acoustic vibration test will be to demonstrate that acoustically generated noise levels more severe than those expected during launch, will not adversely affect or damage the spacecraft structure or the payload instruments.

4.3.4.6 Post-Vibration Deployment Test

The instrument will be subjected to vibration as the result of deployment of some satellite equipment. The deployment tests have four (4) parts: (1) boom deployment, (2) cant deployment, (3) solar array deployment, and (4) antenna deployment.

The instrument will be in the Launch mode for these tests.

4.3.4.7 Final Electrical Check

The final electrical check will be a "Launch Site SEPET". This will be identical to the ambient SEPET.

5.0 NOTES

5.1 Waivers

The following waivers have been granted for the MHS.

Waiver	Date of Approval
NONE	

APPENDIX A. REQUIREMENT DATES FOR MHS INSTRUMENT DATA

			SCR	PDR	CDR
2.1.2	Instrument Contractor				
	Originated Documents				
	Thermal Interface Control Drawing			X(P)	X(F)
	Outline Interface Control Drawing (Mechanical)			X	X
	Electrical Interface Control Drawing				X
	Top Assembly Drawing			X	X
	Bench Check Test Procedure	Delivery of Inst.			
	Bench Check Unit Operation	Delivery of BCU			
	Instrument Handling Procedure				X
	Spec Verification and Calib.	Delivery			
	Data Book	of Inst.			
	Reduce Thermal Model			X	
	Electrical ICD (Logics Diags.)				X
3.1.2.2	Connector Keying Requirements			X	
3.1.3.2.1	Power Dissipation (Table 4)			X	
3.1.3.2.4	Transient Loads (Fig. 4)				X
3.1.3.2.5	DC/DC Converter Frequency		X		
3.1.3.3.1	Power Dissipation				X
3.1.3.3.3	Transient Loads (Fig. 5)			X	
3.1.4.2	Synchronization Signals			X	
3.1.4.3	Commands (Table 5)			X	
3.1.5.2.1	General Requirements			X	X
3.1.5.4.2	Analog Telemetry Points (Table 6)			X	
3.1.6	Test Points				X

NOTE: Where two X's appear for any item, the first X is the date by which preliminary data is required.

APPENDIX A. REQUIREMENT DATES FOR MHS INSTRUMENT DATA (Continued)

		SCR	PDR	CDR
3.2.1.1	Dimensions		X	
3.2.1.3	Moments of Inertia			X
3.2.1.5	Center of Mass			X
3.2.2.1	Instrument Mounting Surface	X		
3.2.5.4	Reference Surfaces		X	
3.2.6.2	Precautions			X
3.2.7	Inst. Material and Finishes		X	X
3.3.1.1	Reduced thermal model and surface model		X	
3.3.3	Temperature Design Limits (Table 9)		X	
3.3.5	Instrument Thermal Control Components		X	
3.4.1	Magnetic Characteristics			X
3.4.3	Flight Environment			X
3.5.1	Storage Requirement (Temp.)		X	
3.5.2	Test Requirements			X
3.5.3.1	Command Sequences			X
3.5.3.2	Turn-On Constraints			X
3.5.3.3	Initial Turn-On Constraints			X
4.0	Instrument Integration, Test and Operating Requirements & Constraints			X
5.1	Waivers*			

* As necessary

APPENDIX B. ATNAGE SUBROUTINES FOR PROCESSING MHS DATA

A. REAL TIME PROCESSING

Many of these functions require the instrument unique Software to be ON. Functions not requiring unique software are marked (*).

1. Raw Data Prints and formatted (Galoppo) prints.

2. Status Checking:

    a. Prints out a status report on commands

    b. Prints out when there is a status bit change. (Operator may inhibit from the keyboard).

3. Scan Verification: The correct position of each antenna in scan mode is calculated. Errors are indicated when actual scan position does not agree with calculated position.

4. Limit Checker: The Limit checker verifies that telemetry functions are within specified bounds. These bounds are established by data base and may be temporarily changed from the keyboard.

    *a. Analog Telemetry functions

    b. Housekeeping Telemetry functions

5. Noise Check: The noise check monitors the instrument noise performance in all five channels by calculating the rms noise levels from the on-board calibration target pixel data.

6. Calibration Check: The calibration check is used during spacecraft thermal-vacuum testing to monitor the instruments gain, offset and noise performance in all five channels.

7. * Command Verification: Housekeeping telemetry status bits which are expected to change when commands are sent are verified by corresponders. The housekeeping telemetry status to be verified is defined in a data base.

B. <u>NON-REAL TIME PROCESSING (OFF-LINE)</u>

TBD